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FLIGHT PLANNING AND CONDUCT OF THE X-15A-2 ENVELOPE EXPANSION PROGRAM

A F F T C

JOHNNY G. ARMSTRONG
Aerospace Research Engineer

Technology Document No. 69-4

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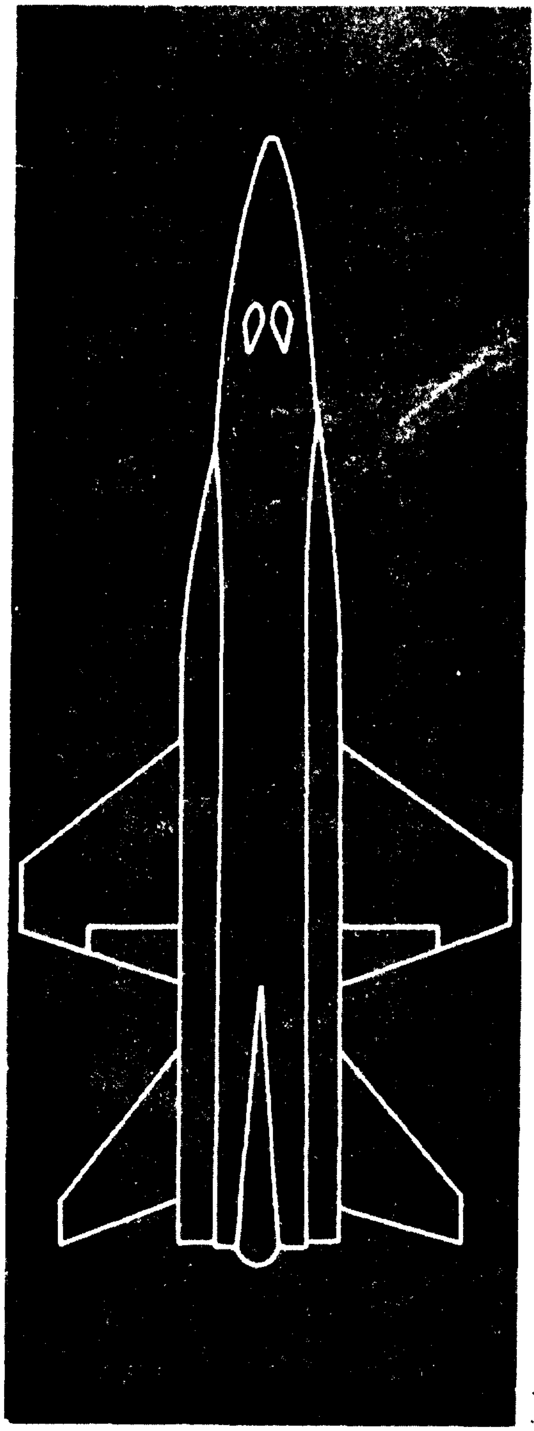

FLIGHT PLANNING AND CONDUCT OF THE X-15A-2 ENVELOPE EXPANSION PROGRAM

JOHNNY G. ARMSTRONG
Aerospace Research Engineer

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FOREWORD

This Technology Document presents a general history of the X-15-2 airplane and a detailed summary of the flight planning and conduct of the test program on the modified X-15A-2. The test program of the modified aircraft was conducted at Edwards AFB, California from 25 June 1964 to 3 October 1967 through the joint efforts of the NASA Flight Research Center and the Air Force Flight Test Center. The aircraft was flown by AFFTC pilots, Colonel Robert A. Rushworth; Major William J. Knight; and Mr. John B. McKay of the NASA Flight Research Center. The actual envelope expansion program consisted of eight flights between November 1965 and October 1967. At publishing time the aircraft was being transferred to the Air Force Museum at Wright-Patterson AFB, Ohio.

The participation of AFFTC personnel in this program was authorized by AFFTC Project Directives 60-82C, dated August 1964 and 67-30, dated 7 September 1966 and was performed under Program Structure 653A 63201F. All photos are courtesy of NASA-FRC, Edwards, California.

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ABSTRACT

After having been extensively damaged during an emergency landing on its thirty-first flight (November 9, 1962), the X-15-2 aircraft was rebuilt and modifications were incorporated to increase the vehicle performance capability to allow flight testing of a hypersonic ramjet engine. The increased performance was derived from additional propellants contained in two external drop tanks. A total of 22 flights were made with the modified aircraft which had been redesignated the X-15A-2. The initial flights were to evaluate the handling quality changes resulting from the modification. The modified propellant system with the external tanks was satisfactorily developed on a ground test stand and performed adequately during flight. Although successful ejection of the external tanks occurred on their separation from the aircraft on each flight, carrying the tanks imposed new constraints on flight planning such as tank ejection flight limits, tank impact locations and revised emergency lake requirements. The ablative material developed to protect the aircraft against temperatures exceeding the original aircraft design appeared to perform satisfactorily on the two fully coated flights flown. On the last flight of this aircraft, the vehicle achieved a maximum Mach number of 6.7 (without using all the propellants available). Extensive heat damage was encountered on the dummy ramjet and lower ventral fin as a result of unexpected increased heating rates due to shock impingement and flow interference effects. While the aircraft was being repaired, the X-15A-2 program was terminated and the maximum speed capability of the aircraft was never achieved. The type of problems encountered during the course of the envelope expansion program may well be expected on other vehicles operating in the speed regime where aerodynamic heating will be an appreciable factor. For instance, the test program was slowed by premature landing gear extensions during flight as a result of aerodynamic heating. These failures should serve as a warning of the potential problems that could occur as a result of minor modifications when operating in a high temperature environment. In addition to the continued demonstration of piloted landing of an unpowered low L/D vehicle, other techniques developed during the program are applicable to orbital lifting re-entry vehicles: application of and flight with an ablative coating, protection of a canopy window with a pilot-actuated covering, and development of an extendable pitot tube as an airspeed source for the terminal landing maneuver.

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LIST OF ABBREVIATIONS AND SYMBOLS

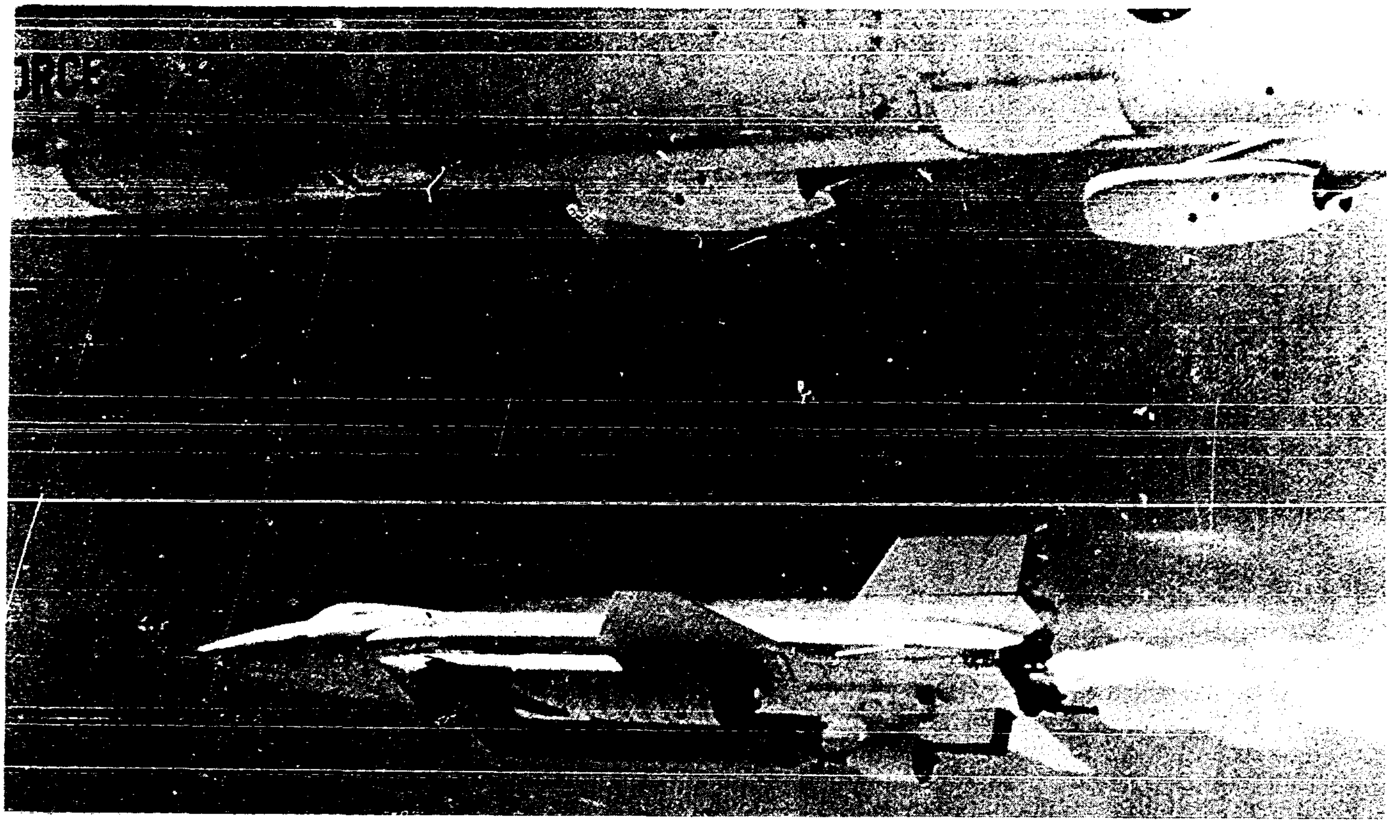
Item	Definition	Units
A_n	normal acceleration	g
A_x	longitudinal acceleration	g
A_y	lateral acceleration	g
AFFTC	Air Force Flight Test Center	- - -
ASAS	alternate stability augmentation system	- - -
C_D	drag coefficient	dimensionless
C_L	lift coefficient	dimensionless
C_{m_α}	change in pitching moment coefficient with change in angle of attack	per deg
C_{n_β}	change in yawing moment coefficient with change in sideslip	per deg
C_{l_β}	change in rolling moment coefficient with change in sideslip	per deg

List of Abbreviations and Symbols Cont'd

Item	Definition	Units
$C_{n_{\delta_r}}$	change in yawing moment coefficient with change in rudder position	per deg
$C_{l_{\delta_r}}$	change in rolling moment coefficient with change in rudder position	per deg
cg	center of gravity	percent MAC
fps	feet per second	- - -
H_2O_2	hydrogen peroxide	- - -
I_x	moment of inertia about longitudinal body axis	slug-ft^2
I_y	moment of inertia about lateral body axis	slug-ft^2
I_z	moment of inertia about vertical body axis	slug-ft^2
L/D	ratio of lift to drag	dimensionless
LOX	liquid oxygen	- - -
NH_3	anhydrous ammonia	- - -
NM	nautical miles	- - -
$\dot{p}$	roll angular acceleration	rad/sec^2
psf	pounds per square foot	- - -
PSTS	propulsion system test stand	- - -
$\bar{q}$	dynamic pressure	lb/ft^2
$\dot{q}$	pitch angular acceleration	rad/sec^2
$\dot{r}$	yaw angular acceleration	rad/sec^2
SAS	stability augmentation system (3 axis rate dampers)	- - -
W	vehicle weight	lb
$\bar{X}$	longitudinal cg change	ft
$\bar{Y}$	lateral cg change	ft
$\bar{Z}$	vertical cg change	ft
δ_h	horizontal stabilizer position	deg
δ_r	rudder position	deg
α	angle of attack	deg

X-15 Flight Identification   Flight 2-53-97

2	refers to #2 X-15
53	number of launches
97	number of times airborne on B-52



INTRODUCTION

The X-15A-2 obtained a maximum Mach number of 6.7 on October 3, 1967. At the time this flight occurred the X-15A-2 was involved in an envelope expansion program to extend the maximum Mach number capability from Mach 6 to approximately Mach 8. The aircraft was then to be used as a flying testbed for testing a hypersonic ramjet engine. Financial cutbacks following this flight resulted in termination of X-15A-2 from the active flight program without ever having realized the aircraft's maximum velocity capability.

● GENERAL AIRCRAFT HISTORY

The X-15 free flight program began with a glide flight of X-15-1 on June 8, 1959. X-15-2 entered the flight program on September 17, 1959, making the second X-15 flight and the first powered flight with the XLR11 engine. Two XLR11 engines were used to power each of these two aircraft to begin the envelope expansion program of the X-15 before the XLR99 engine became available. The results of the envelope expansion program with this interim engine are documented in reference 1. The X-15-2 made a total of nine flights with this engine configuration and flew the first flight with the XLR99 engine on November 15, 1960. During 1961 the aircraft was involved in the envelope expansion program with the XLR99 engine, making a total of nine flights that year and obtaining a maximum altitude of 217,000 feet and a maximum Mach number of 6.04.

Ten flights were made with the X-15-2 in 1962 with the majority of the flights being designed to obtain aerodynamic heating data at the high Mach number and high dynamic pressure under quasi-steady conditions.

During the latter part of 1962, flight tests were conducted to verify a predicted improvement in lateral directional handling qualities at high Mach numbers and high angles of attack with the lower movable ventral fin removed. The thirty-first flight of X-15-2 on November 9, 1962, was planned to further investigate the aircraft's ventral-off handling qualities. However, during this flight only 30-percent thrust could be obtained from the XLR99 engine as a result of a throttle control failure. An emergency landing was attempted at the launch lake (Mud Lake) in accordance with preplanned alternate procedures. At touchdown the left main gear strut collapsed, causing the aircraft to skid sideways and turn over on its back. The gear structural limit was exceeded primarily because landing flaps failed to extend. The aircraft suffered extensive damage (figure 1). The aircraft had accumulated a total free flight time of 4 hours, 40 minutes, 32.2 seconds at this point.

A decision was made to rebuild the aircraft in order to complete the planned experiments and to incorporate modifications to increase the performance of the aircraft and thus allow it to be used as a testbed for a hypersonic ramjet engine. Approval was given under Contract AF33(657)-11614 for North American Aviation, on May 13, 1963, to proceed with repair and modification of the aircraft at a cost of approximately 5 million dollars. The modified aircraft (figures 2 and 3) was returned to Edwards AFB on February 19, 1964, and made its first flight June 25, 1964.

Figure 1 AIRCRAFT DAMAGE – FLIGHT No. 2-31-52

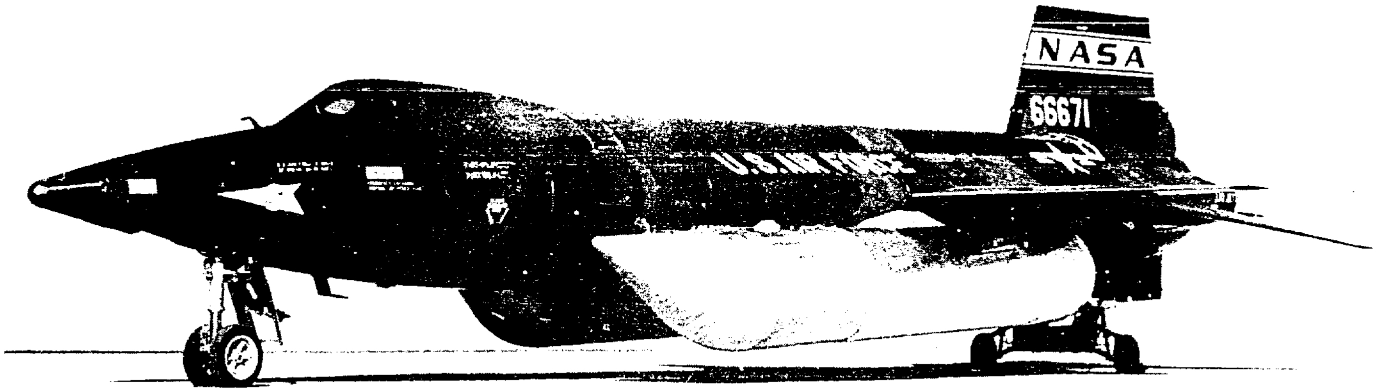


Figure 2 AIRCRAFT AFTER REPAIR AND MODIFICATION – ¾ FRONT

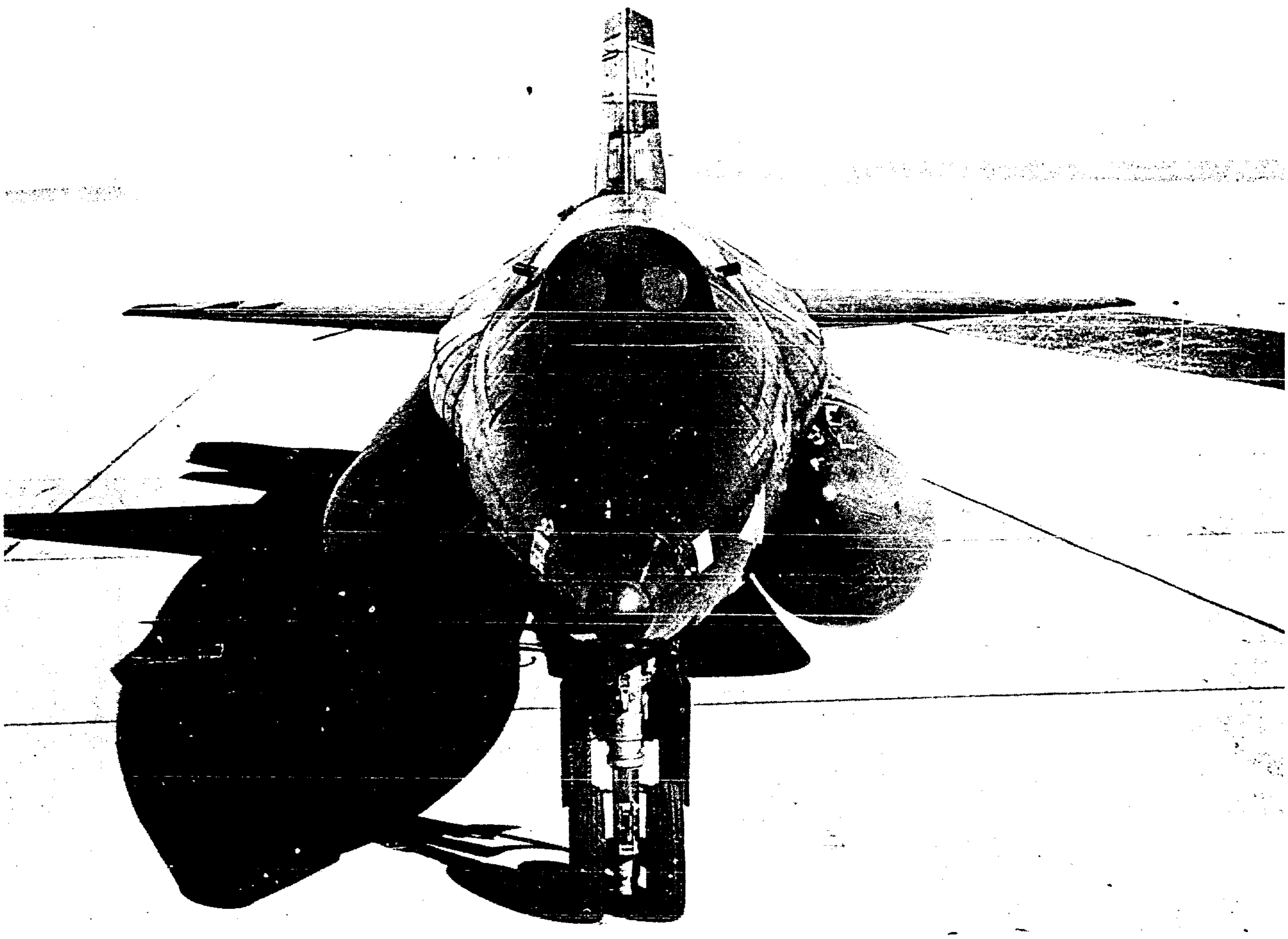


Figure 3 AIRCRAFT AFTER REPAIR AND MODIFICATION – FRONT

DESCRIPTION OF AIRCRAFT

Numerous changes were made to the X-15-2 while it was being rebuilt. The major change was the addition of two external propellant tanks. These jettisonable external propellant tanks were designed to increase the engine burn time by approximately 70 percent, thereby increasing the performance capability of the aircraft required for testing the ramjet engine. The external tanks were each approximately 23.5 feet long and 38 inches in diameter.

The left-hand tank (figure 4), weighing 1150 pounds empty, contained three helium bottles required for propellant tank pressurization in addition to a capacity for approximately 793 gallons of LOX. The right-hand tank (figure 5) weighed 648 pounds empty and contained approximately 1080 gallons of anhydrous ammonia. The total weight of additional propellant to be carried in the external tanks was approximately 13,500 pounds. Because of the difference in empty weight and in propellant volumes, the left-hand tank was approximately 2000 pounds heavier than the right at launch.

The external tank jettison system (figure 6) contained two sets of fore and aft gas cartridges to eject the tanks from the aircraft. In addition, the design included a solid propellant sustainer rocket on the nose of each tank to impart a nose-down moment upon jettison to improve separation characteristics at supersonic speeds. For a normal empty tank jettison both sets of gas cartridges were fired and the nose rocket was ignited. In the case of a requirement to make an emergency tank ejection while the tanks were still full, only one set of the gas cartridge ejectors were fired and the nose rocket was not activated.

The high cost of these tanks dictated that they be reusable, hence each tank contained its own recovery system consisting of a drogue and descent chute. The drogue chute was deployed immediately after separation and the main descent chute deployment was initiated by a barometric sensor normally set for 8,000 feet.

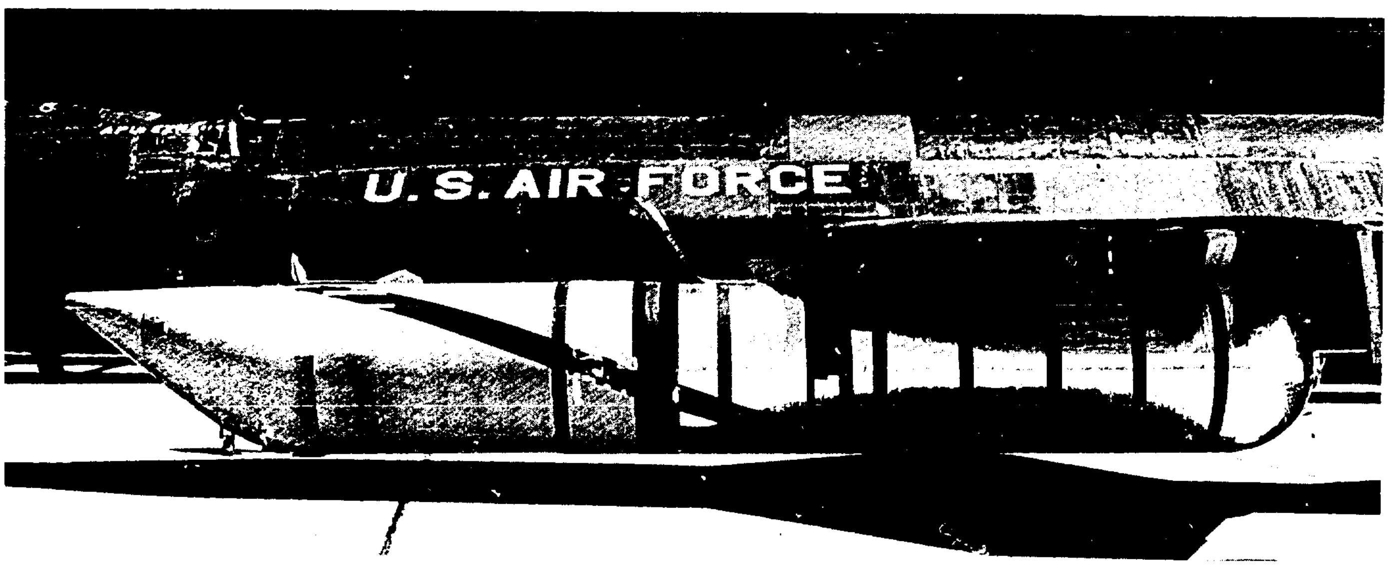


Figure 4 LOX TANK

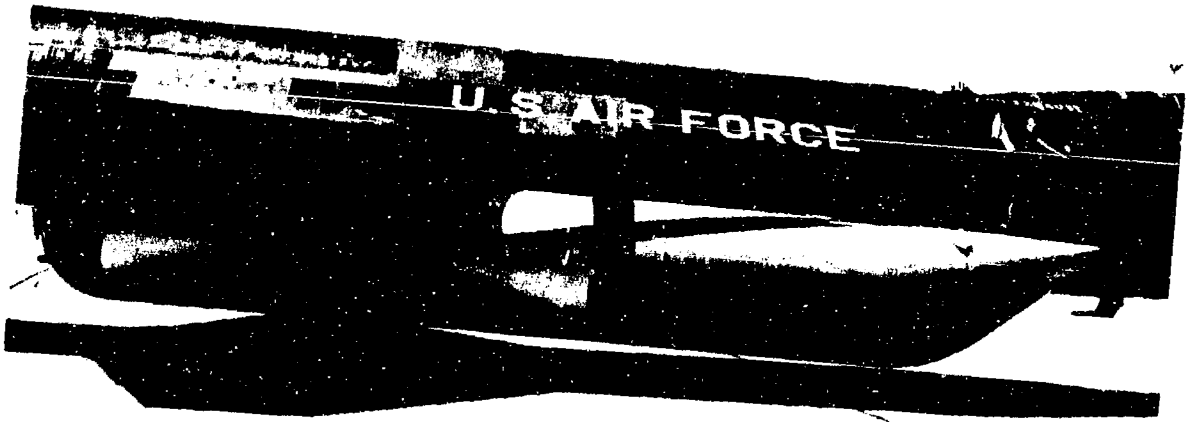


Figure 5 ANHYDROUS AMMONIA TANK

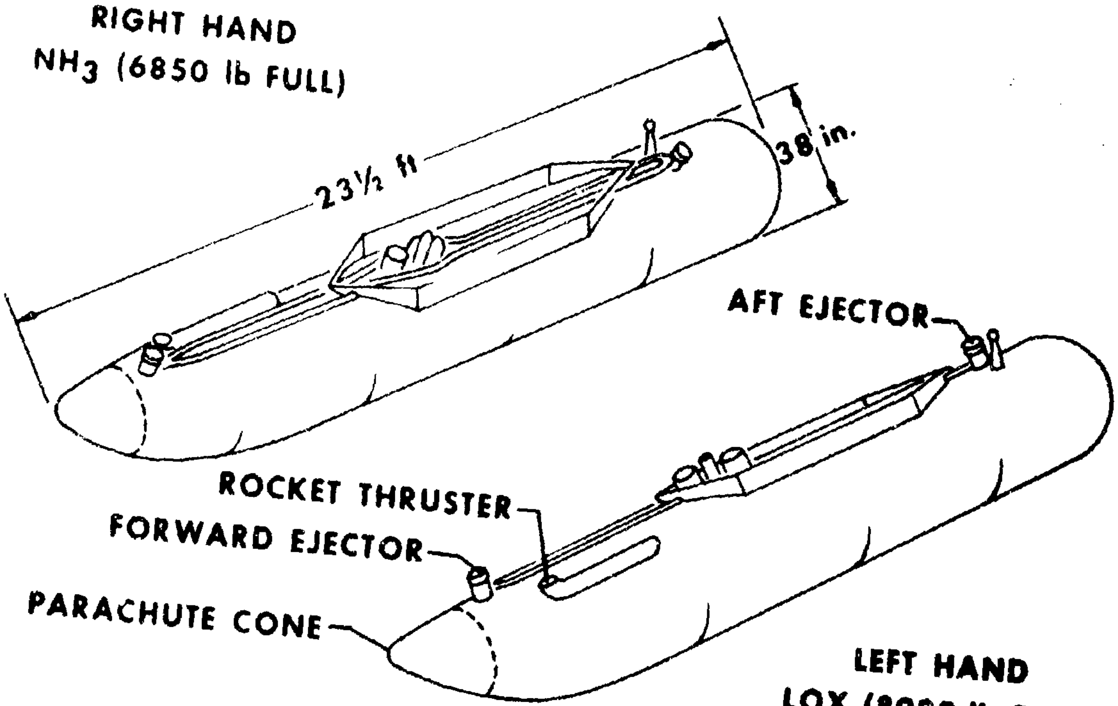


Figure 6 EXTERNAL TANK CONFIGURATION

Other modifications to the basic X-15-2 airframe are shown in figure 7. A 29-inch extension was added to the fuselage in the area of the center of gravity between the LOX tank and the anhydrous ammonia tank. Tanks containing 48 pounds of liquid hydrogen for the ramjet engine were to have been installed in this area.

Additional hydrogen peroxide required for the extended engine propellant pump operation was stored in tanks in the extended aft side fairings. A helium tank for additional propellant pressurization gas was installed on the aft fuselage above the engine (figure 7).

The design included a longer landing gear that would provide ground clearance for landing with a ramjet engine installed (shown hypothetically in figure 7). Since the ramjet engine was not to be available until much later in the test program it was decided to take advantage of the increased landing load margin that could result from a shorter main gear during the initial portion of the test program. The strut of this interim gear was 6.75 inches longer than the standard X-15 gear.

Drawing from the experience of the initial X-15 envelope expansion program when the standard windshield design suffered several glass fractures caused by thermal stress near the corners of the rectangular glass retainer, the X-15A-2 windshield was designed with an elliptical shape. In addition, three panes of glass were installed in the new design instead of 2 panes as in the normal X-15.

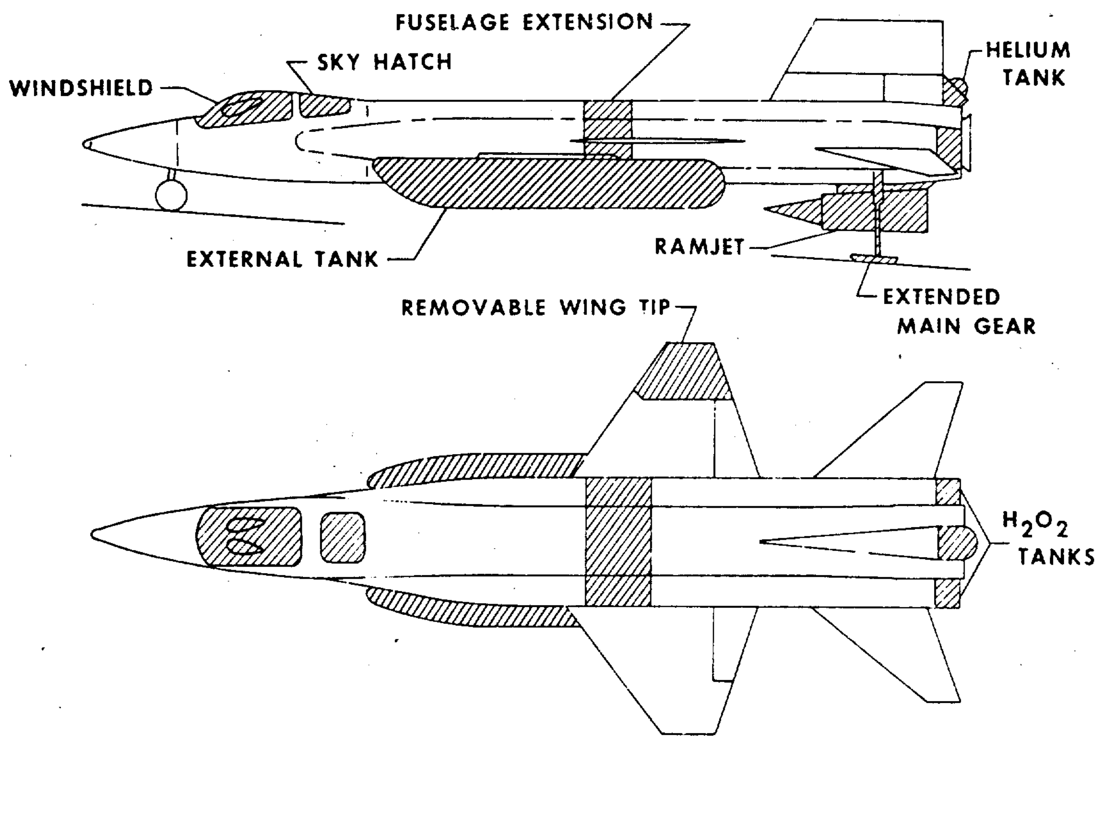


Figure 7 X-15-2 MODIFICATION

To protect the aircraft structure from the high aerodynamic heating in the Mach 6 to 8 regime, an ablative material was chosen to cover the aircraft.

To facilitate projected experiments, two additional modifications were made to the aircraft. A "sky hatch" was added just behind the cockpit which featured doors that could be opened upon command from the pilot near peak altitude on high-altitude flights. The ultraviolet stellar photography experiment (Star Tracker) was later installed in this compartment. The right-hand wing tip was designed to be removable. This removable wing tip was to have allowed testing of advanced materials and/or structural design.

The onboard instrumentation recorders utilized during the envelope expansion program consisted of five 36-channel oscillographs, eight 3-channel oscillographs, two 14-track tape recorders, one 24-cell manometer recorder and one cockpit camera. In addition, an 86-channel PDM telemetry system was used to transmit parameters in real time from the aircraft.

TEST AND EVALUATION

SUMMARY OF INITIAL FLIGHTS AFTER MODIFICATION

Wind-tunnel tests of X-15A-2 were conducted in the summer and fall of 1963. The tests indicated that very little difference existed in aerodynamic characteristics between the modified X-15 without external tanks and the standard X-15. Figure 8 is a comparison of the static stability (C_{m_α}, C_{n_β}) and dihedral effect (C_{ℓ_β}) of the standard and modified X-15 with the lower ventral off at two angles of attack. The movement of the normal flight cg 10 percent forward apparently compensated for the destabilizing effect of extending the fuselage 29 inches forward and thus resulted in little change to the static stability derivatives (C_{m_α}, C_{n_β}). The low level of directional stability of the standard X-15 at Mach 3 and 12 degrees α was slightly lower for the modified aircraft. However, at reentry conditions of Mach 5 and 20 degrees α the dihedral effect still remained favorable (i.e., negative C_{ℓ_β}).

The longitudinal trim characteristics of the modified aircraft remained the same as for the standard X-15 for Mach numbers less than 4 and stabilizer deflections less than 15 degrees. At higher Mach numbers (figure 9) the modified aircraft's trim capability was about 5 degrees less in angle of attack.

The initial flights of X-15A-2 were planned to obtain stability and control maneuvers to verify the wind-tunnel predictions. Data obtained on the flights did in fact verify the wind-tunnel results; however, the verification program took longer than expected when trouble was encountered with the modified landing gear system.

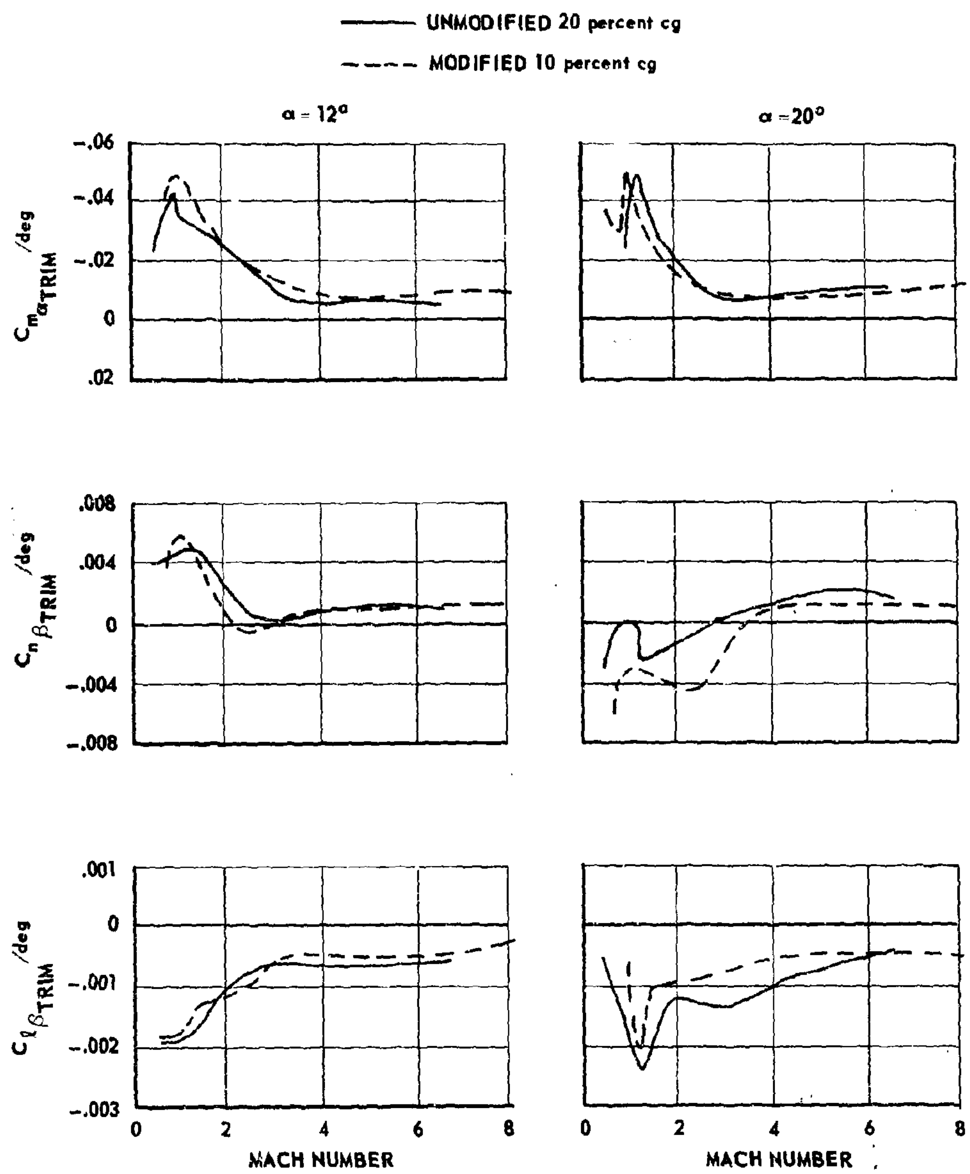


Figure 8 COMPARISON OF STATIC STABILITY AND DIHEDRAL EFFECT BETWEEN THE MODIFIED AND UNMODIFIED X-15 (VENTRAL OFF)

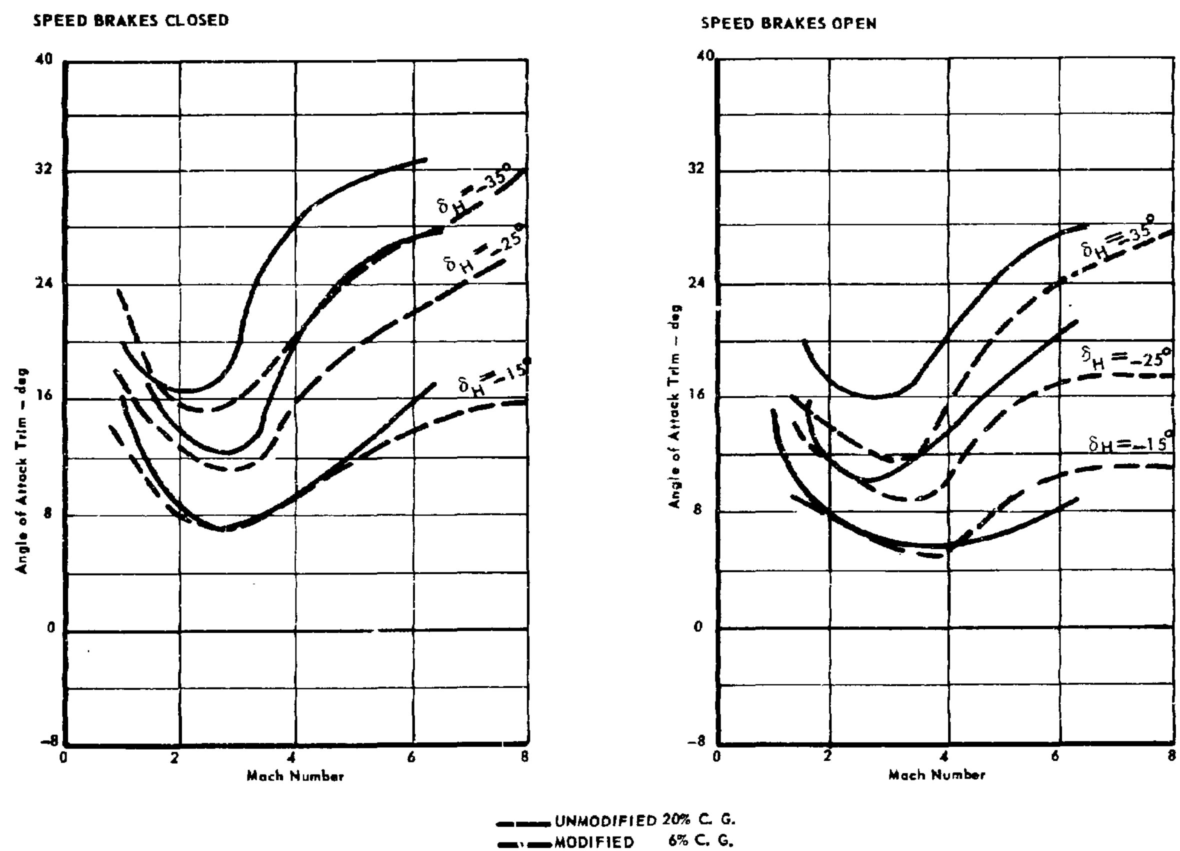


Figure 9. COMPARISON OF LONGITUDINAL TRIM CHARACTERISTICS BETWEEN THE MODIFIED AND UNMODIFIED X-15(VENTRAL OFF.)

On the second flight (2-33-56) of the modified aircraft, after obtaining a maximum Mach number of 5.23, the nose gear inadvertently extended at Mach 4.4. Despite the degraded control and increased drag resulting from the extended gear, the pilot was able to return to Rogers Dry Lake at Edwards Air Force Base. The chase aircraft pilot was able to verify that the nose gear appeared to be structurally sound and in the locked position but that the tires showed heat damage. The pilot elected to attempt a landing which was accomplished normally except that both nose gear tires blew out on landing. Investigation revealed that aerodynamic heating was the cause of the failure; specifically, the expansion of the fuselage was greater than the capacity of the tension regulator/temperature compensator device of the gear release cable. This caused an effective pull on the release cable which then applied a load on the uplock hook. An additional load on the uplock hook was imposed by an outward bowing of the nose gear door. The load from both of these sources caused the uplock hook to bend, allowing the gear to extend. This failure was duplicated by ground tests simulating the fuselage expansion and by applying heat to the nose gear door. The key linkages were redesigned and the system was subjected to the same ground heat test without failure. The same mission plan for stability and control data was planned for the next flight (2-34-57). Again, shortly after shutting down the engine at a maximum Mach number of 5.2, the pilot experienced a similar noise and aircraft trim change at Mach 4.5. The small nose gear scoop door had extended. This door in the normal gear extension sequence was used to impose airloads on the nose gear door to assist in the extension of the nose gear. Although not as serious a failure as that of the previous flight, it again precluded obtaining dampers off stability data. The nose gear door was redesigned to provide positive retention of the scoop door regardless of the thermal stresses.

A slower speed flight (2-35-60) was flown next to a maximum Mach number of 4.66 to check out the modifications on the nose gear door. The nose gear performed normally and additional stability data were obtained.

On the next flight (2-36-63) the right main gear extended at Mach 4.4. Again the chase pilot was able to verify that the gear appeared structurally sound and a normal landing was made. Postflight inspection revealed that the uplock hook had bent allowing the gear to deploy. Again, the source of the high load on the uplock hook was concluded to be from aerodynamic heating. Referring to figure 10, the temperature gradient between the inside and outside of the stowed gear and strut resulted in differential expansion causing the gear to bow in the middle. The additional length of the interim gear caused its thermal load to be almost twice as high as that of the standard gear (figure 11). Thus when the critical temperature was reached in flight, the load became sufficiently large to deform the hook causing the gear to extend. The uplock hook was redesigned and pertinent instrumentation was added to allow flight evaluation of the change. To safely test the modification in flight, a "temperature profile" was selected with heating rates similar to the flight when the failure occurred but a total temperature less than that at which the failure occurred. The normal procedure used to predict the temperature on the vehicle was to fly a profile on the X-15 analog simulator and then load the pertinent data into a digital program that calculated temperature at selected points on the vehicle. To obtain a discrete temperature versus time profile would be an iteration process

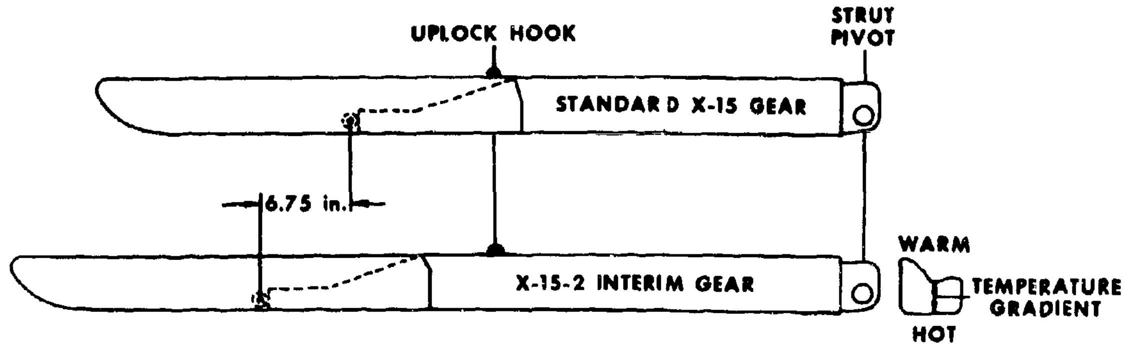


Figure 10 COMPARISON OF MAIN-GEAR GEOMETRY

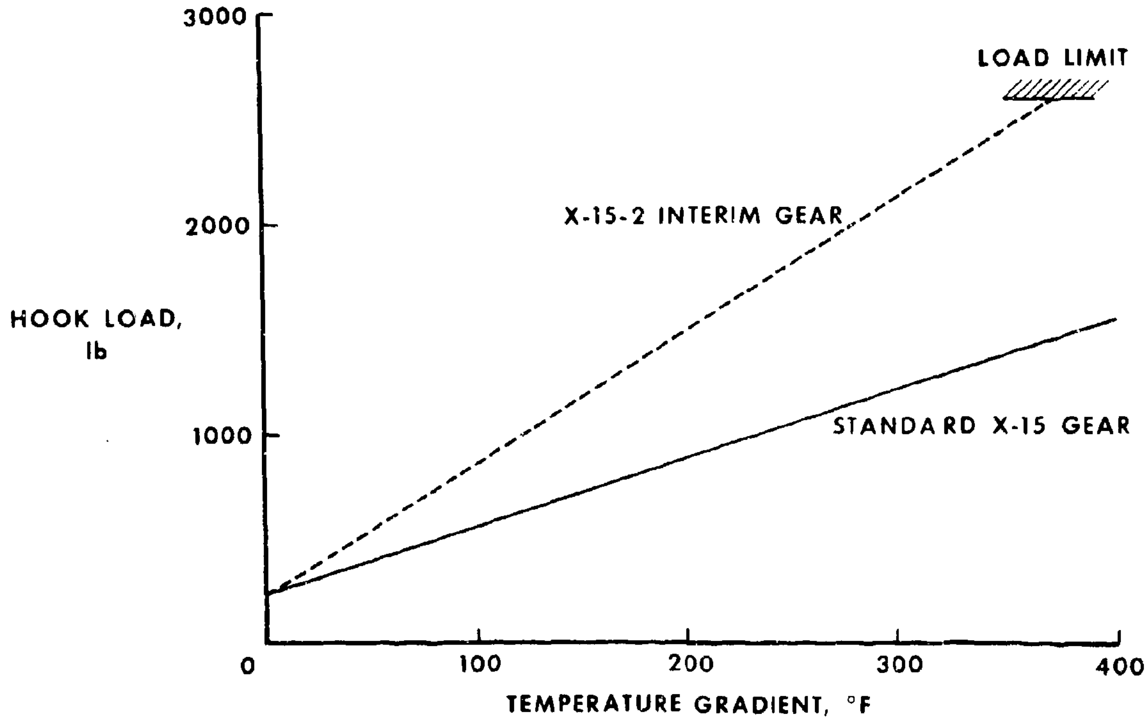


Figure 11 EFFECT OF TEMPERATURE GRADIENT ON UPLOCK-HOOK LOAD.

that could take weeks to accomplish. However, the use of a real time temperature simulation of the X-15A-2 (reference 2) made the task relatively simple. A plot of the desired temperature time history was placed on an X-Y plotter and the flight planner was able to observe the temperature as he simulated the "flight" to assess the match and make immediate adjustments to the flight path until the desired profile was obtained. The practicality of the resulting mission was also evaluated by simulating off-design flights to determine which flight conditions could result in undesirable temperature overshoots.

Five more flights (2-38-66 through 2-42-74) were flown before the envelope expansion program was begun. These flights continued the study of stability and control and landing gear performance tests. Three of the flights were primarily to obtain data for the ultraviolet Star Tracker experiment. However, little usable star tracking data were obtained because of problems incurred in maintaining the precise attitudes required for the experiment.

During successive flights, attempts were made to improve the reaction augmentation system (RAS) which provided rate damping about all three axes with the reaction control system to assist the pilot in maintaining the required aircraft attitudes. The Star Tracker flights were discontinued because of the position of the desired stars at that time of the year.

PREPARATION FOR ENVELOPE EXPANSION FLIGHTS

Prior to the arrival of the modified aircraft and concurrent with the initial flight phase, studies were made of the unique problems associated with flying the aircraft with external tanks.

External Tank Impact Area

The addition of external tanks to the X-15 added an additional constraint to the flight planning task, that of having the aircraft over a satisfactory tank impact area at the time of planned tank ejection. To define the area of probable tank impact, trajectory calculations were made using the following conditions as the standard for tank ejection (table I).

It was assumed in these calculations that the tanks would fly at zero angle of attack, i.e., zero lift. Drag coefficients were estimated for the tanks and for the 33.2 sq ft drogue chute. A wind drift allowance was included while descending with the main chute deployed in the direction of the predominate winds for this area.

This study defined a ground recovery area of 8.1 NM by 9 NM. The legal clearance to drop the tanks in the geographic area defined by this study had to be obtained.

Additional areas were established for a failure of the drogue chute and for an emergency tank ejection immediately after X-15 launch. Although it was not feasible to obtain land rights to drop tanks in the entire area so defined because of the large land area involved, the possibility of an impact in these areas was definitely considered in

TABLE I

EXTERNAL TANK TRAJECTORY

PARAMETERS

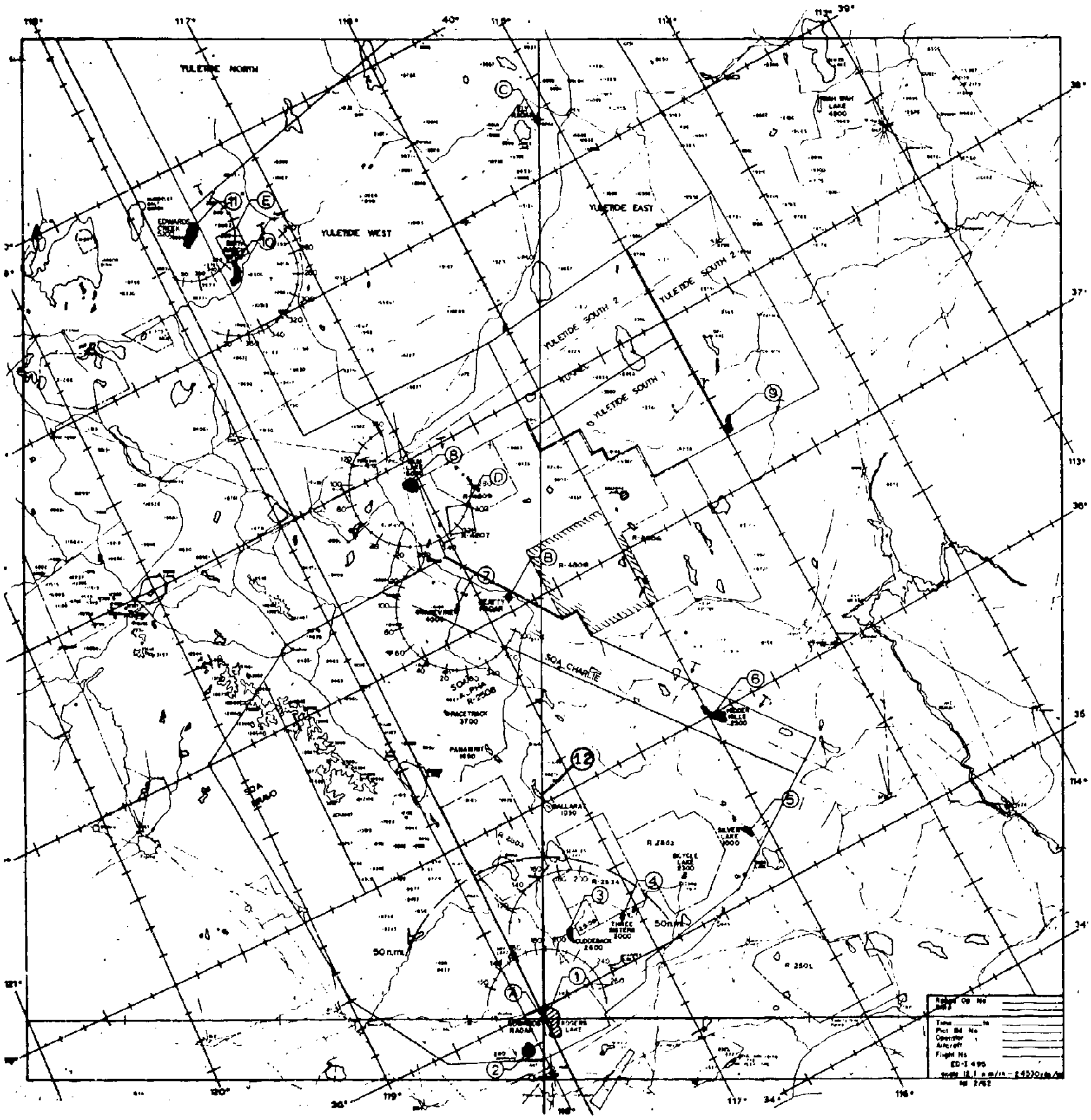
ASSUMED CONDITIONS	DEVIATIONS
Velocity-2040 fps	Velocity ± 150 fps
Altitude-65,000 ft	Altitude ± 5000 ft
Flightpath Angle 30 degrees	Flightpath Angle ± 2.5°
Distance from X-15 Launch-15 N.M.	Down range launch point error ±1.N.M.
	Cross range launch point error ± 1.5 N.M.
	Launch heading error ±2.5°

selecting the ground track of the X-15 with external tanks installed. The approach taken was similar to that of operational aircraft flying with external stores, namely, that a drop at an unplanned location would be the result of some malfunction or emergency.

Emergency Lake Coverage

The X-15 flights were planned with the requirement that the aircraft always be within gliding distance of a dry lake suitable for landing. Thus the X-15 was launched within gliding distance of a "launch lake" and during its flight back to Edwards passed by several dry lake beds that had been tested and marked with runways for X-15 landing (figure 12). During the entire X-15 program, 10 landings were made at these remote lake beds.

Since the initial acceleration of the X-15A-2 with external tanks was considerably less than that of the standard X-15, it was necessary to re-evaluate the emergency lake coverage for flights with external tanks. A parametric simulator study was made to determine the glide capability of the aircraft for different engine burn-time along the design profile to 100,000 feet. A summary plot of this study is shown in figure 13. Placing the geometry of the existing emergency lakes at their respective positions on the distance scale makes possible a quick analysis of the emergency lake coverage available. This analysis concluded that, of the existing launch points, only a launch from Mud Lake was



1 ROGERS DRY LAKE	10 SMITH RANCH DRY LAKE
2 ROSAMOND DRY LAKE	11 EDWARDS CREEK VALLEY DRY LAKE
3 CUDDEBACK DRY LAKE	12 BALLARAT DRY LAKE
4 THREE SISTERS DRY LAKE	A EDWARDS RADAR
5 SILVER DRY LAKE	B BEATTY RADAR
6 HIDDEN HILLS DRY LAKE	C ELY RADAR
7 GRAPEVINE DRY LAKE	D TANK IMPACT AREA
8 MUD DRY LAKE	E TANK IMPACT AREA
9 DELAMAR DRY LAKE	LAUNCH POINTS

Figure 12 X-15 EMERGENCY LAKES & HIGH RANGE

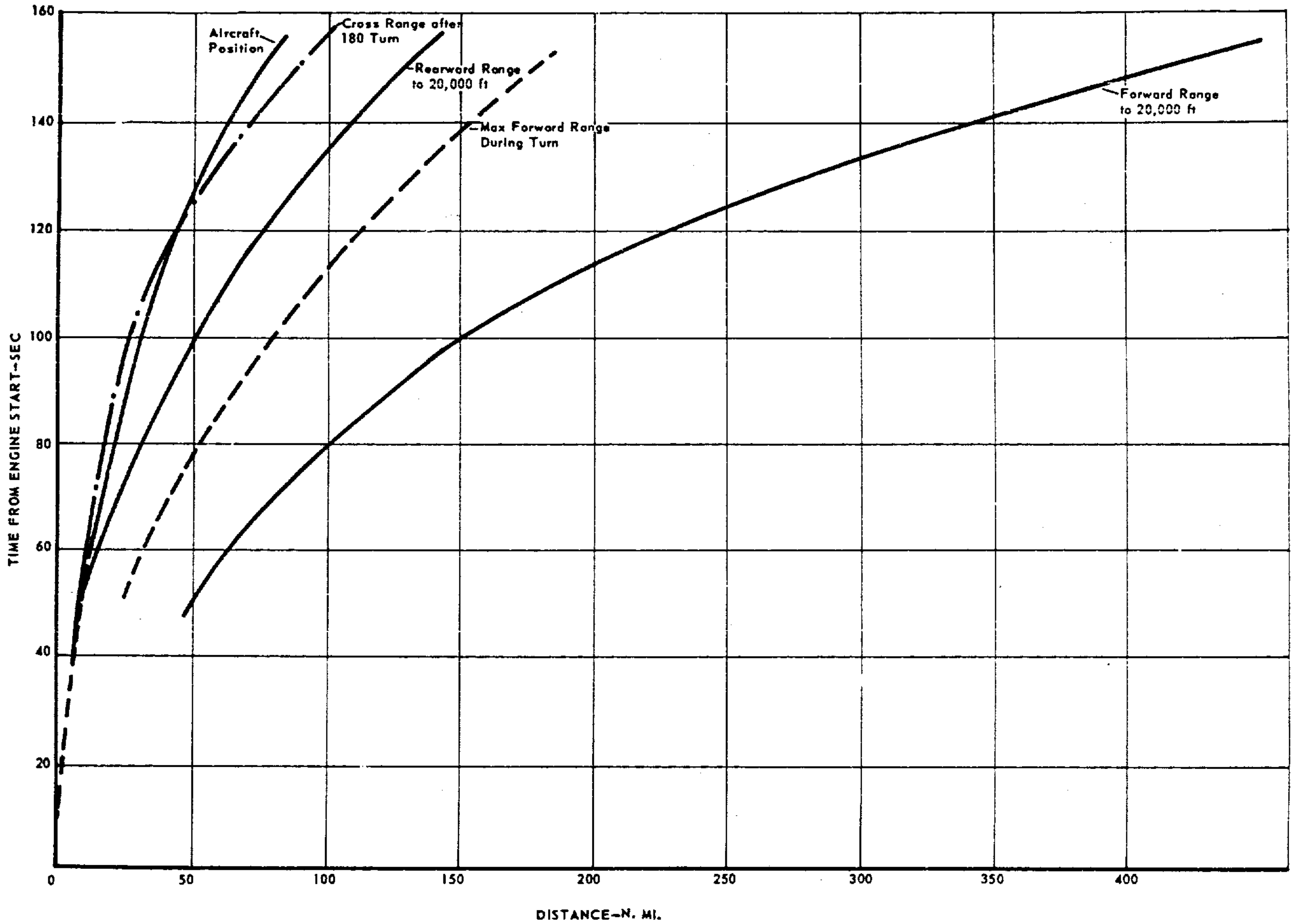


Figure 13. X-15A-2 MACH 8 DESIGN MISSION RANGE CAPABILITY vs TIME

suitable for flight with external tanks. However, since the Mud Lake launch point is only 187 nautical miles from Edwards, a more distant launch point would have been required for flights with maximum velocity approaching 7000 fps. The use of Smith Ranch as a launch point was desired but unfortunately the distance between Smith Ranch and Mud Lake was too great for the glide capability of the aircraft and a time period existed when the aircraft would have been without a suitable landing site It was hoped that a usable lake could be found between Smith Ranch and Mud Lake to fill the time gap that existed. An uprange survey of dry lakes yielded no such usable landing site. However, a relatively large dry lake, Edwards Creek Valley, approximately 15 NM northwest of Smith Ranch was found to be suitable. The emergency lake coverage from an Edwards Creek Valley launch, although not as good as desired (at least a 20,000-foot high key), did provide for a straight-in approach to Smith Ranch and Mud Lake if an engine shutdown occurred at the most critical time. In addition, for an emergency occurring at the time of tank ejection, a landing could have been performed at Smith Ranch. Upon completion of the study that proved Edwards Creek Valley to be suitable and required for use as part of the envelope expansion flight program, considerable coordination was required to obtain the right to use the lake as an emergency landing site and to obtain approval to drop the tanks in a specified area near Smith Ranch.

External Tank Seperation

The utilization of external tanks on the X-15 was unique in that the tanks had to be ejected from the aircraft. Structural limitations of the aluminum tanks and degrading handling qualities dictated that the maximum allowable Mach number with the external tanks be 2.6. Prior to reaching that speed the tanks had to have separated from the aircraft cleanly. A recontact with the aircraft could have possibly resulted in immediate catastrophic failure or apparent minor local damage that could later become catastrophic as high temperatures were encountered. A normal landing with the tanks installed was not possible because of the increased drag and lack of ground clearance. Hence, considerable effort was expended to assure adequate separation characteristics of the tanks from the aircraft.

Theoretical analyses were made of the separation characteristics based on force and moment wind-tunnel data obtained with the tanks in the vicinity of the X-15 model. Dynamic tank ejections were also made in the wind tunnel. Good agreement between the two methods of analysis were obtained (reference 3). The velocity and pitch rate imparted to the tanks by the ejector system were determined from qualification tests. Based on these tests and analysis, the ejection boundary shown in figure 14 was established; between +10 and -2 degrees angle of attack and at dynamic pressures less than 400 psf. However, simulator studies showed that the major portion of the planned profile from launch to depletion of the external propellants (figure 14) was outside the allowable ejection boundary and that precise control would be required to achieve the satisfactory ejection conditions. A re-analysis of the data indicated that acceptable, although not as good, separation characteristics would probably exist at dynamic pressures up to 600 psf and that this increased boundary could be verified from results of the initial tank ejections.

Prior to the first tank flight, two dummy tank ejection tests were performed from the aircraft. The aircraft was placed over a 10-foot deep pit (figure 15). Beams with similar mass and inertia properties were constructed to simulate the empty tanks. Preloaded cables attached to the beams applied simulated aerodynamic drag and side loads. A single set of ejector cartridges was used on the first test at simulated air loads of 400 psf dynamic pressure, 5 degrees angle of attack and 3 degrees sideslip. The second test used both sets of ejector cartridges at a simulated dynamic pressure of 600 psf. Both tests were successful and high-speed motion pictures showed good separation characteristics. During the tests the hydraulic and electrical power was supplied by the aircraft and the SAS system was engaged to assure that no detrimental effects on the aircraft system occurred during the simulated tank ejection.

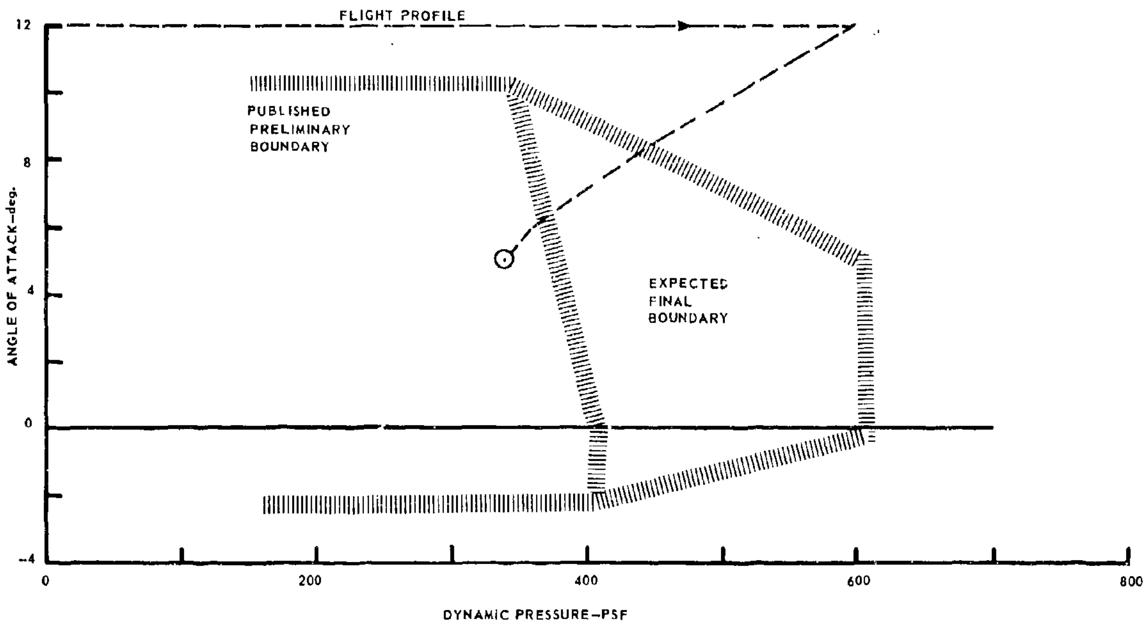


Figure 14 EXTERNAL TANK EJECTION BOUNDARY

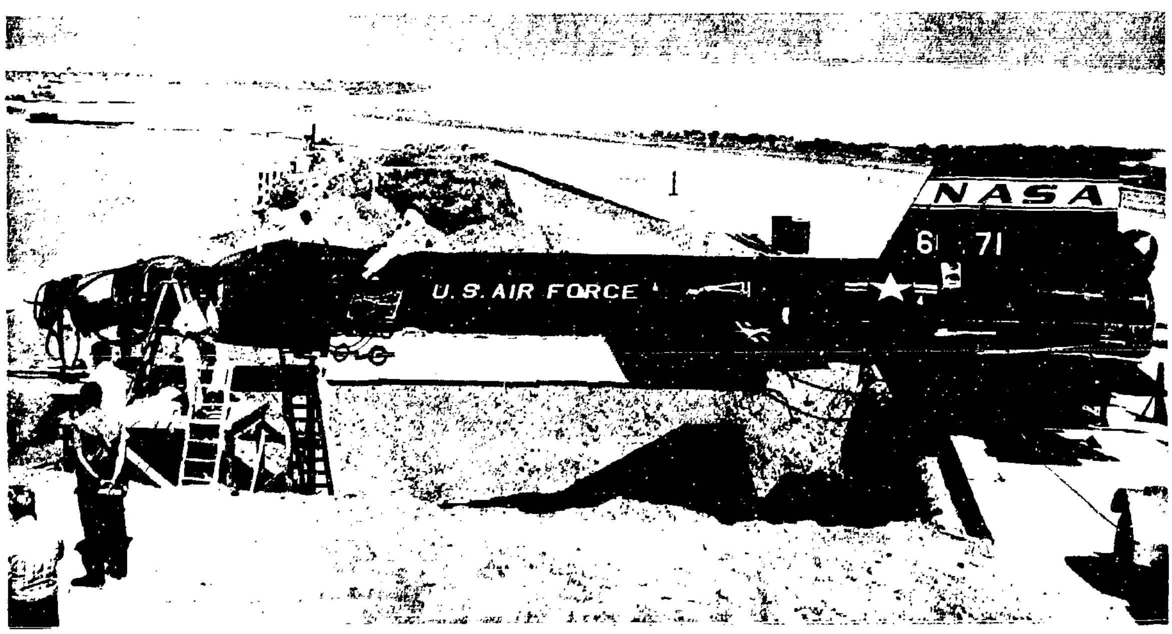


Figure 15 DUMMY TANK EJECTION TEST SETUP

During a Design and Operating Criteria Review of the modified aircraft, concern was expressed for the separation characteristics of the external tanks with the tanks partially full. The pilot could have found himself in such an emergency situation requiring the ejection of the tanks with a partial load of propellant if the engine shut down prematurely within the first 60 seconds of flight. The tanks and the ejection system were designed for only a full or empty tank ejection. It was considered that the tanks with the ejection system as initially designed would not withstand the loads imposed during ejection with partial fuel. Studies were initiated to find a suitable solution to this possible problem area. Three separate approaches were studied as follows:

1. A rapid external propellant dump system that would empty the external tanks in 15 seconds.

2. A system of tank baffles that would reduce fuel slosh.

3. A rapid fill system that would allow the external tanks to be filled from the internal tanks.

Each of these schemes had its own advantages and disadvantages, but all complicated the system design further and required excessive time before the completed hardware could be designed, constructed and qualified for flight. After much study and consideration it was decided that the flight program should be begun with the tanks as initially designed. This calculated risk was in part considered reasonable because to that date the XLR99 engine had not encountered a premature shutdown from 100-percent thrust after a successful light was obtained after launch. However, the study did bring forth one design change to the tank ejection system that was incorporated. A third ejection button was added to the cockpit for ejection of both external tanks with a partial load of propellant. In order to reduce the loads imposed on the tanks at ejection, the new button activated only one set of the ejection cartridges and also caused the separation nose rocket to fire.

Handling Qualities

The X-15 simulator was updated with the wind-tunnel determined derivatives of the X-15A-2 with the lower ventral installed and with external tanks installed. A complete assessment of the predicted handling qualities of the aircraft in this configuration was performed on the simulator.

A lower level of longitudinal static stability existed with the external tanks installed making the aircraft considerably more sensitive to control. This control sensitivity may be seen in figure 16 which compares the stabilizer required for changes in angle of attack with and without tanks at Mach 0.9 and 2.0. Note that full stabilizer trim was required to maintain a trim angle of attack of 12 degrees without external tanks and less than 1/3 of full trim for the same angle of attack with the external tanks on.

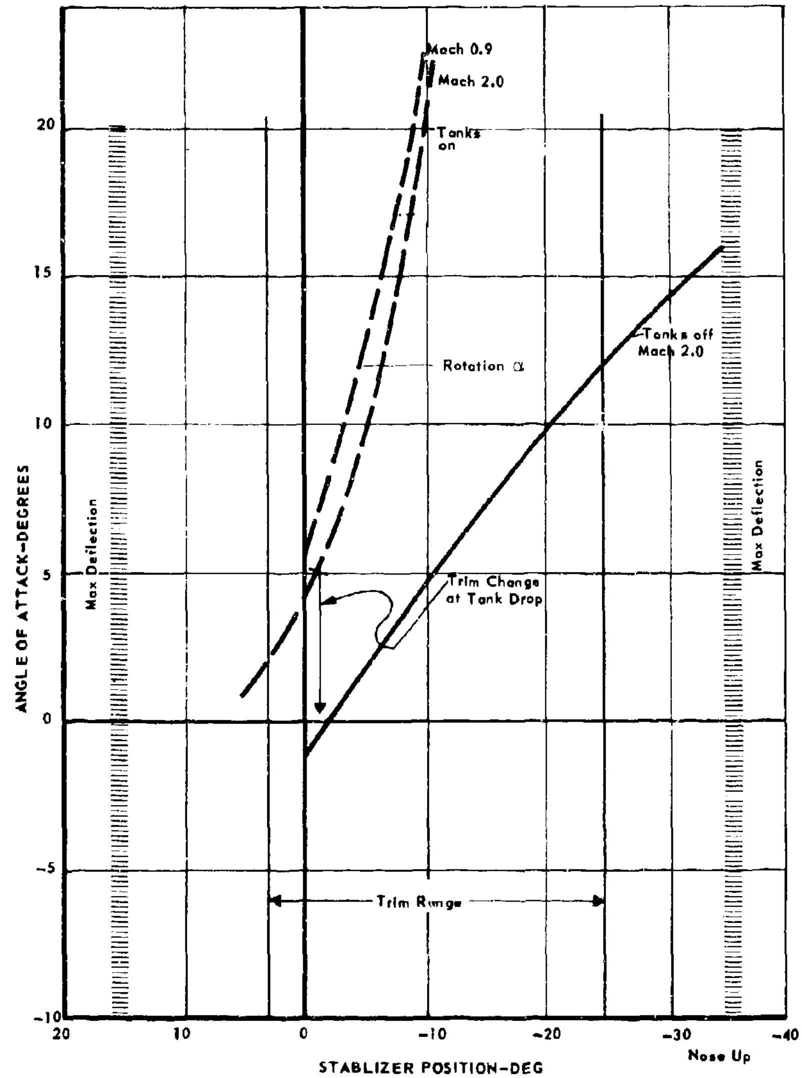


Figure 16 EXTERNAL TANKS ON & OFF LONGITUDINAL TRIM COMPARISON

The overall control task was further complicated by the offset center of gravity caused by the external tanks. The center of gravity variations with the external tanks installed is shown in figure 17 as a function of engine burning time. At launch, the vertical cg was approximately 9 inches below the aircraft center line, and became less as the external propellants were consumed. This offset below the thrust vector resulted in a nose-down pitch at engine light that had to be counteracted with additional nose-up stabilizer trim. The left lateral displacement of the cg caused by the heavier LOX tank and propellants, resulted in a left rolling moment which had to be counteracted by the pilot with right aileron.

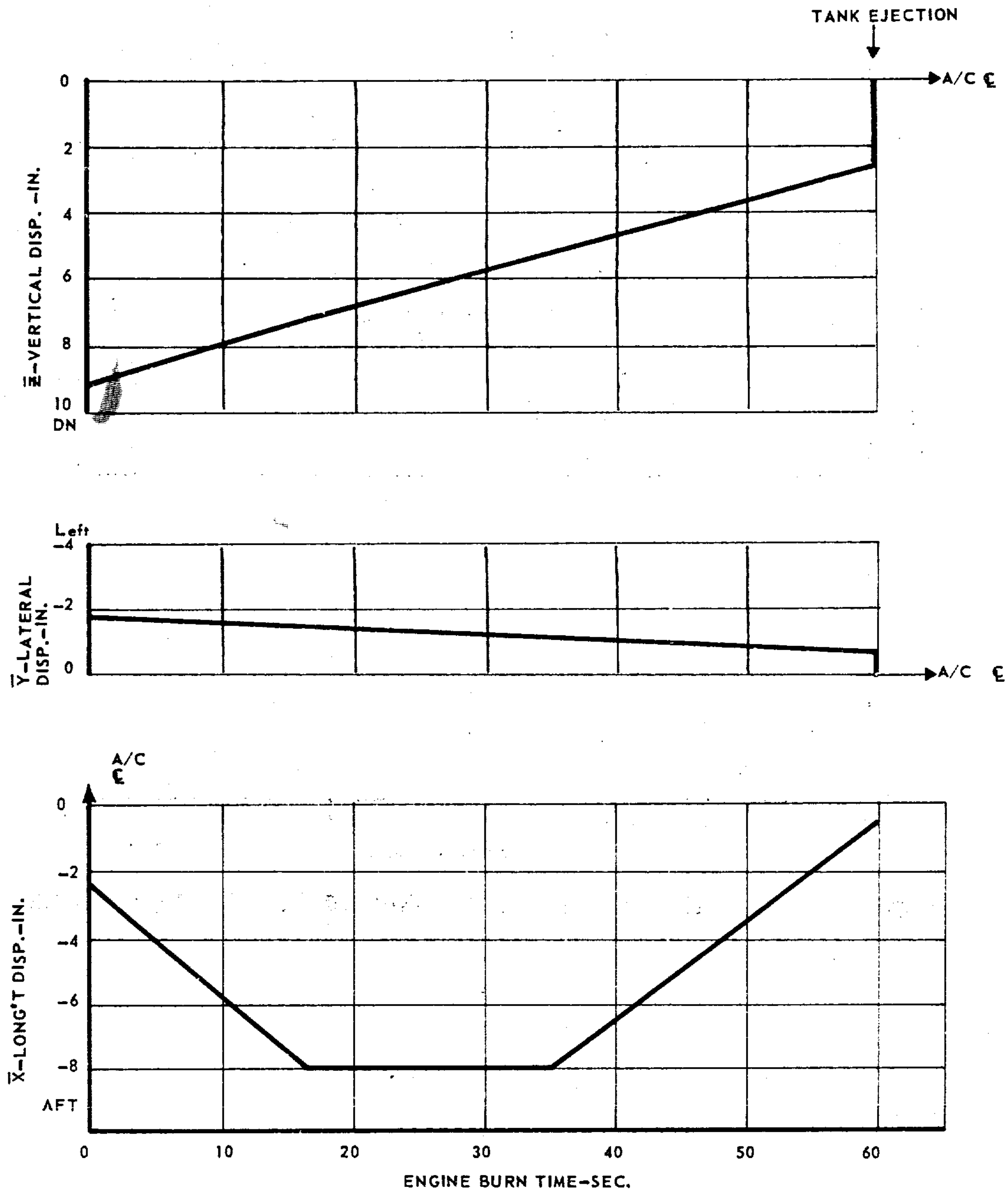


Figure 17. X-15A-2 cg VARIATION WITH EXTERNAL TANKS ON

The offset cg was accounted for in the simulator by summing the change in angular acceleration defined by the following equations into the basic equations of motion.

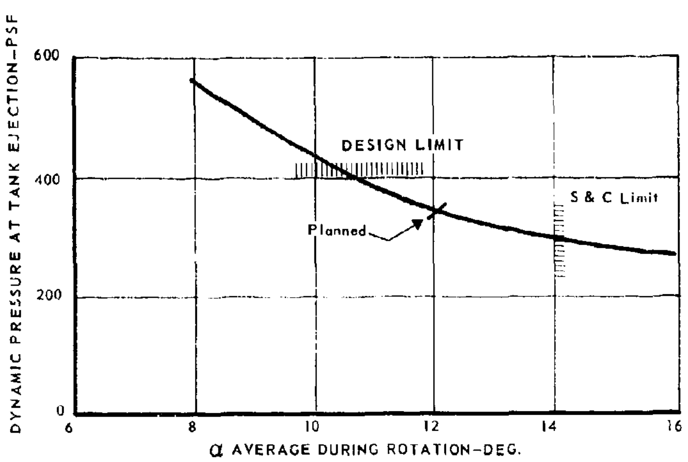
$$\Delta\dot{p} = \frac{\overline{Y}W}{I_x}(a_n) - \frac{\overline{Z}W}{I_x}(a_y)$$

$$\Delta\dot{q} = -\frac{\overline{X}W}{I_y}(a_n) - \frac{\overline{Z}W}{I_y}(a_x)$$

$$\Delta\dot{r} = -\frac{\overline{X}W}{I_z}(a_y) + \frac{\overline{Y}W}{I_z}(a_x)$$

The increased handling qualities task with the external tanks on was of concern in light of the predicted restrictions imposed by the flight limitations. The maximum usable angle of attack was felt to be 14 to 15 degrees because of the low static directional stability and the aileron control required to counteract the lateral offset cg. Other limits that restricted the usable angle of attack were the maximum allowable dynamic pressure of 1000 psf with the external tanks on, and the requirement to be less than 400 psf at tank drop. The results of a simulator parametric study show the effect of these limits (figure 18). The α during rotation after launch had to be greater than 8 degrees to keep $\bar{q}$ less than 1000 psf. In order for the aircraft to be at less than 400 psf when the external tanks were ready to be ejected required an angle of attack during the rotation of at least 10 degrees. Therefore, it was established that the aircraft would be flown at 12 degrees α with only a ± 2 degrees angle of attack error allowable during the rotation to the desired pitch angle which was normally 35 degrees. This pitch angle was then maintained until 5 seconds before the planned tank drop time when a pushover to 5 degrees α was accomplished. Five degrees α was chosen as the condition for tank ejection based on the expected aircraft trim change when the tanks separated. Referring to figure 16, the expected trim change due to the difference in stabilizer effectiveness was a 5-degree nose-down pitch; hence, by ejecting the tanks at 5 degrees α, the aircraft would pitch down to zero α ($\approx$zero g).

Simulator studies predicted the handling qualities with the ventral on (without external tanks) to be good with the dampers operative. However, with the roll and/or yaw dampers inoperative the handling qualities were considerably degraded. Figure 19 shows the areas (Mach vs α) of predicted negative damping.



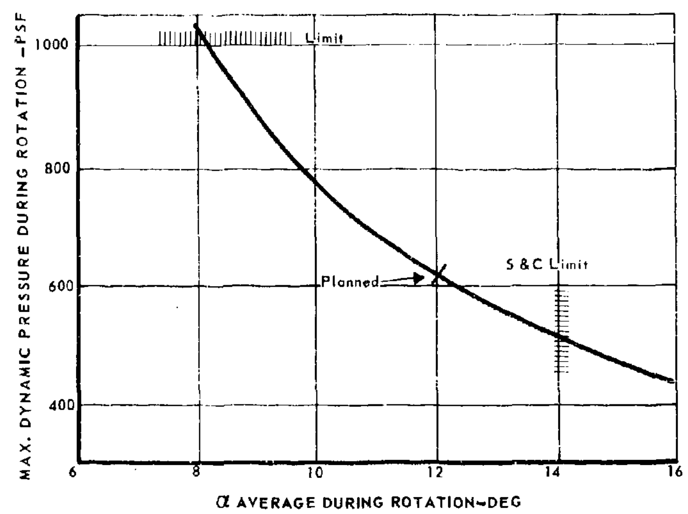


Figure 18 EFFECT OF ROTATION ANGLE OF ATTACK ON LIMITING DYNAMIC PRESSURES.

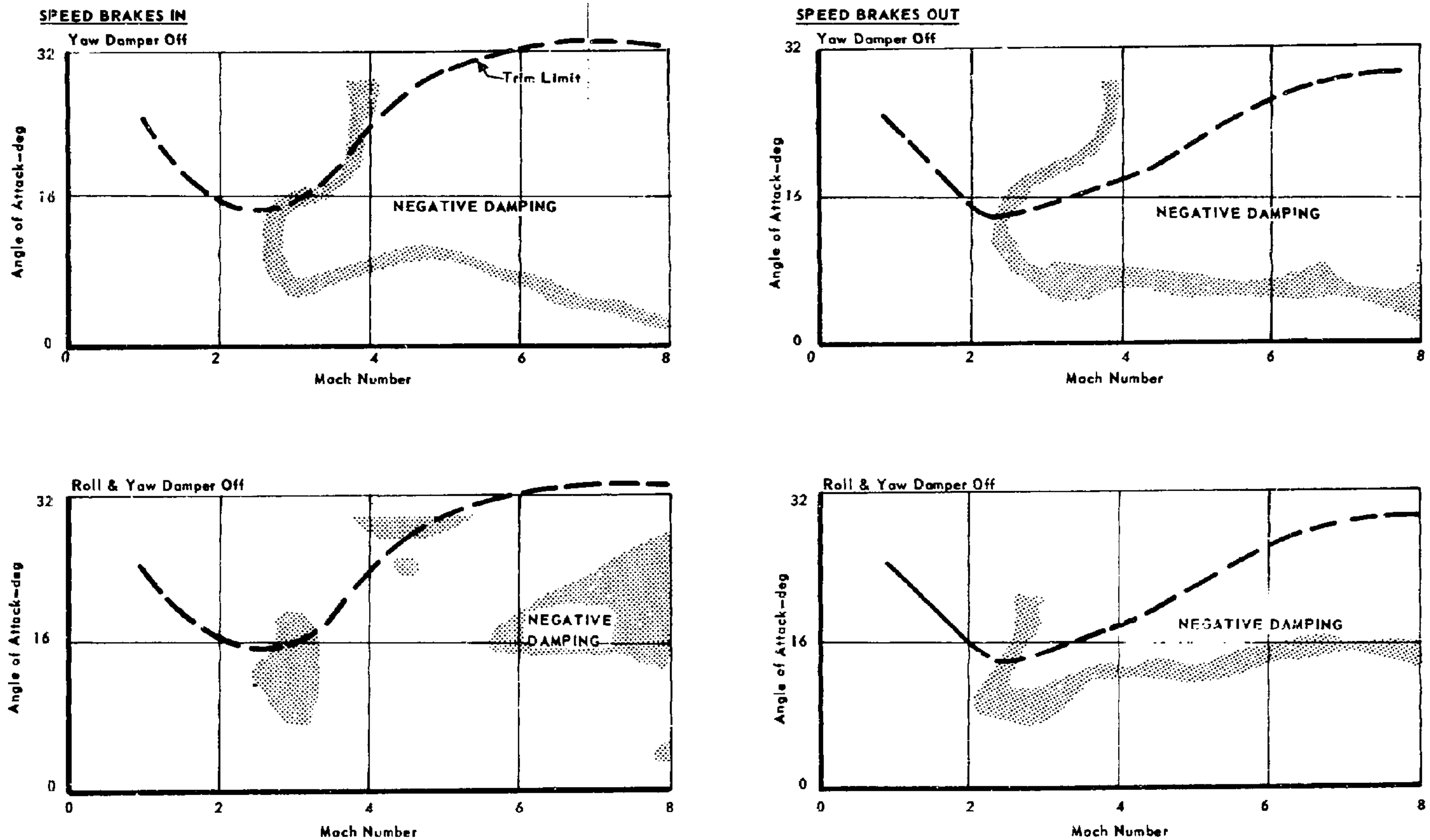


Figure 19. X-15A-2 STABILITY BOUNDARIES Ventral On

As was the case with the basic aircraft, the poor handling qualities at the high angles of attack was due primarily to the large negative dihedral effect (positive C_{ℓ_β}) caused by the presence of the lower ventral fin. For a yaw damper failure with the speed brakes out (figure 19), a divergent sideslip oscillation persisted above about 6 degrees angle of attack. Although the divergence could be damped by the pilot with rudder inputs, continuous attention to the task was required. The simulator also showed that the divergent yaw oscillation could be eliminated by turning off the roll damper, however, the pilot would then have to accept the task of flying the aircraft with less lateral-directional stability. From the simulator studies it was determined that, because of the relatively low altitude profiles required, the aircraft could be safely flown after a roll and/or yaw damper failure by maintaining an angle of attack of less than 8 degrees. For the initial envelope expansion flights, this characteristic could be accepted and attempts would be made to obtain flight verification. For the projected ramjet tests, where flight at high dynamic pressure would be required, a divergence of this type could have been too rapid for the pilot to take corrective action. Hence, it was deemed desirable to provide a redundant yaw damper, and design of an alternate yaw damper similar to that existing in the pitch and roll axis was begun.

External Tank Propellant System

The design philosophy of the advanced propellant system was to retain, as much as possible, the same hardware and fluid flow passages of the existing system. The propellants in the external tanks were fed into the existing three compartmented internal propellant tanks as the internal propellants were consumed.

To test the design and component performance of this propellant system, the existing Propulsion System Test Stand (PSTS) was modified to duplicate the aircraft. This approach had the following obvious advantages:

1. Eliminated the hazards to the aircraft and personnel.

2. Allowed the aircraft to proceed with "tanks-off" flight program.

3. Allowed the installation of more instrumentation than would have been possible on the aircraft.

During the tests, several component deficiencies and design discrepancies were encountered and corrected (reference 4). A successful full duration run for 135.5 seconds was made at 100 percent thrust.

To qualify the aircraft's system, an engine run was made with the external tanks installed on the aircraft prior to the first flight with full external tanks.

Ablatives

The initial design of the modified aircraft included an ablative material to protect the aircraft structure. This material was later considered unacceptable for use on the X-15. The principal objections to the original material were a cure-cycle requiring a heat of 300 de-

grees F, a water solubility problem, and poor thermal protection efficiency.

In late 1963 a joint NASA-USAF committee was formed to select an ablative material suitable for the X-15A-2 application. Initially a large number of materials were screened with respect to the following areas: shielding effectiveness, room-temperature cure-cycle, bond integrity, operational compatibility and refurbishment. The total weight of the thermal protection system was not to exceed 400 pounds. The initial testing of candidate materals was accomplished in the 2-inch arc jet at the University of Dayton Research Institute. Flight tests of sample materials at different locations on the X-15's were valuable in uncovering differences in the materials under actual flight environment; particularly in terms of application techniques, bonding effectiveness, and resistance to aerodynamic shear loads. Final evaluation was accomplished at the NASA Langley Research Center's 2500 KW arc jet under heating conditions simulating peak heating rates expected on the X-15A-2 maximum velocity mission. Four ablative materials qualified for the X-15A-2 application. A request for proposal was sent to the manufacturers of the materials. In late 1965 a contract was let to the Martin-Marietta Company to design and apply a sprayable silicone ablator to the aircraft. The basic ablative material was designated MA-25S and had a virgin material density of 28 pounds per cubic foot. The material could be sprayed and cured on the aircraft at room temperature (70 to 100 degrees F). Special premolded fiber reinforced silicone material (ESA3560-IIA) similar to that used on the Air Force PRIME[1] vehicle was designed for all leading edges. A premolded flexible material (MA-25S-1) was developed to cover seams of access panels required for preflight activities. This ablative material as well as all the other candidate materials was known to be impact sensitive in the presence of LOX. Tests showed that a local detonation would occur on the material submerged in LOX when struck with a force of 8.5 foot-pounds. Special precautions were taken to prevent contamination of the aircraft systems by the ablative material. The interior of the aircraft was protected during application by masking off all openings into the aircraft and contamination measuring devices were installed in the interior to verify the protection. Filters were installed in the propellant line for inflight protection. An ablative sealer (DC90-090) was applied over the final coat to prevent flaking off of the ablative material during maintenance and pre-flight preparation. In addition, this rubbery white sealer decreased the LOX sensitivity to 26 foot-pounds. A more detailed description of the ablative material; properties, application and performance is contained in reference 6.

Canopy Eyelid

During the arc tests it was observed that loosened material from the ablative surface tended to reattach to surfaces downstream of the test specimen. Flight tests were performed with a panel of X-15 windshield glass mounted on the vertical tail aft of a sample patch of the ablator.

1 PRIME (Precision Recovery Including Maneuvering Entry) was the designation of the development and reentry flight test of the SV-5 Lifting Body under the Spacecraft Technology and Advanced Reentry Test (START) Program. Program 68A, Program Element 63409874.

The glass panel opaqued, which could have restricted the pilot's vision (figure 20). Three different concepts were considered to protect the canopy windshield:

1. Explosive fragmentation of the outer windshield glass.

2. Boundary layer blowing over the windshield area.

3. Hinged metal shield (eyelid).

The eyelid was chosen as the most practical method and the design was incorporated onto the aircraft's left windshield (figure 21). The eyelid was to be closed prior to launch and not to be opened again until the aircraft approached the landing site at speeds less than Mach 3.

Pitot-Static System

The standard pitot-static pickups had to be relocated and redesigned because of the presence of the ablative material. The standard static source was located on the side of the forward fuselage which would be surrounded by ablative material. A vented compartment behind the canopy was chosen as a static location and found to be suitable during flight tests. The standard dog-leg pitot tube ahead of the canopy was to be replaced by an extendable pitot (figure 22) since temperatures above design would have been experienced at high Mach numbers. This tube remained within the fuselage until the aircraft decelerated below Mach 2. The pilot then actuated the release mechanism and the tube extended into the airstream.

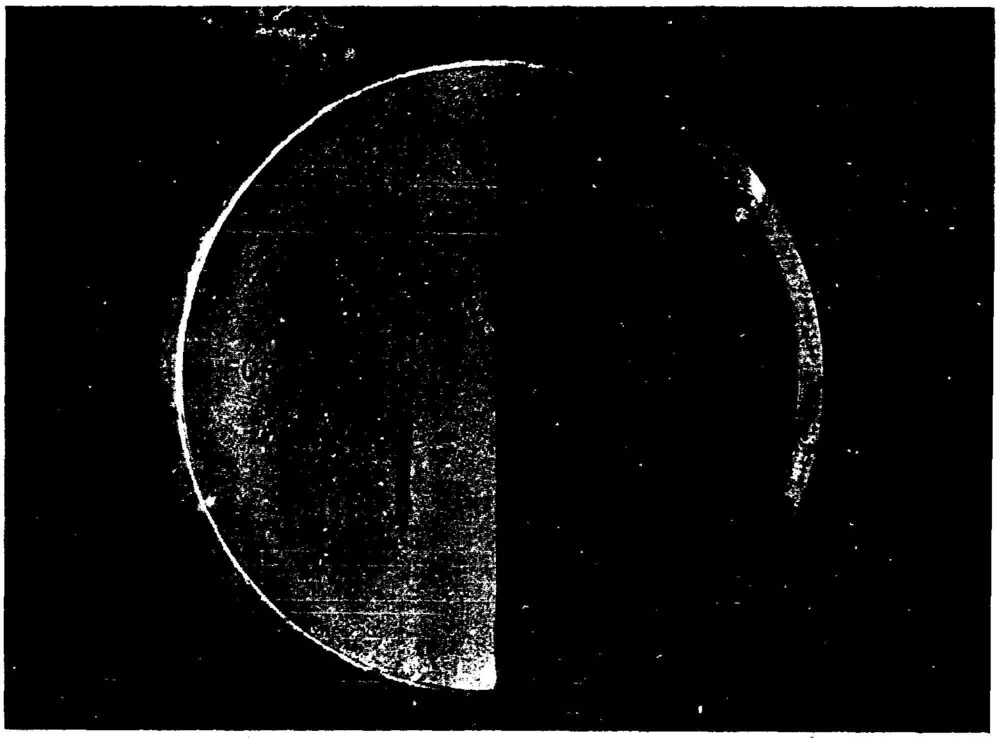


Figure 20 ABLATIVE PRODUCTS ON CANOPY GLASS TEST PANEL

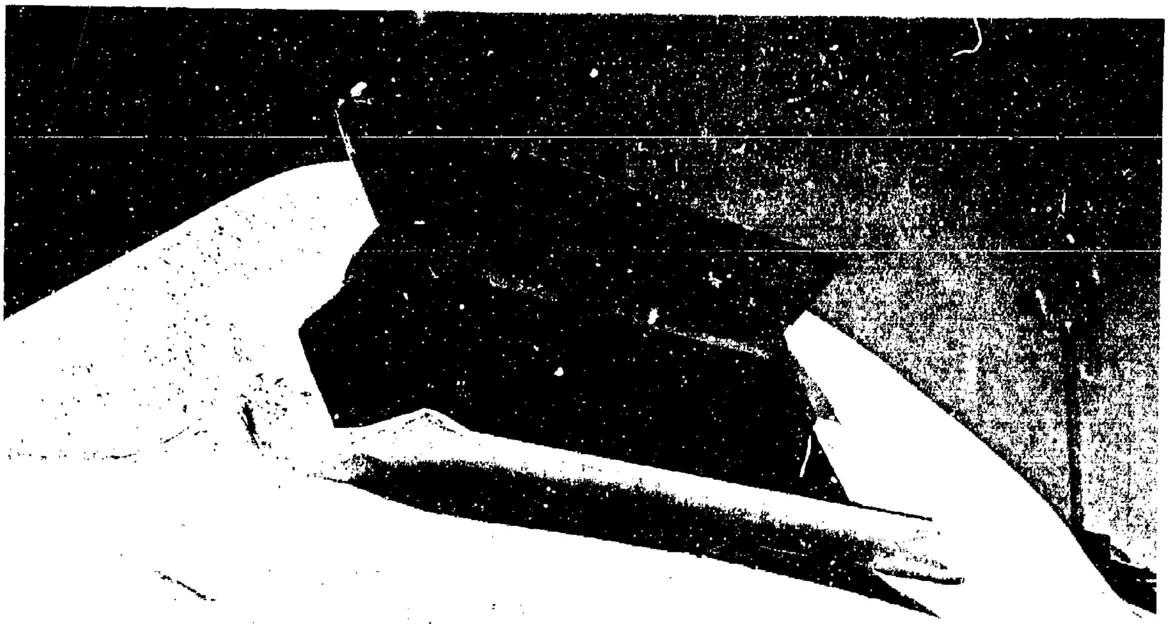


Figure 21 CANOPY EYELID

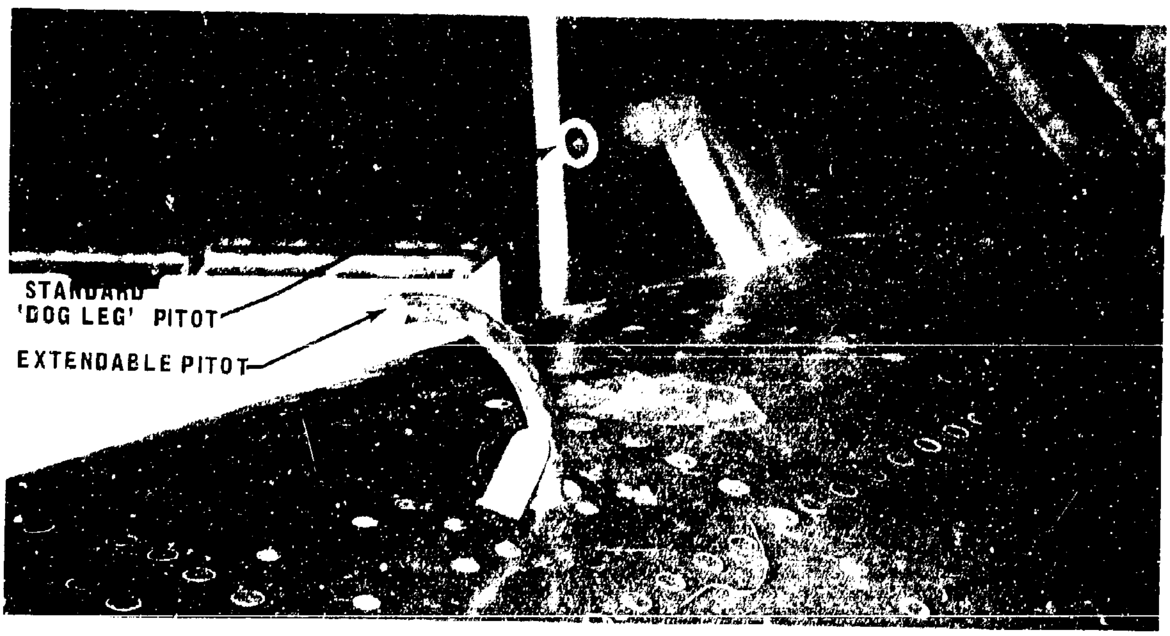


Figure 22 PITOT TUBES

ENVELOPE EXPANSION FLIGHTS

The envelope expansion of the X-15A-2 was accomplished on eight flights between November 1965 and October 1967. Each of the flights will be described in detail in order to provide a better understanding of the step-by-step approach utilized in expanding the envelope and the problems that were encountered along the way.

The aircraft configuration for the flights are summarized in table II.

Table II

SUMMARY OF AIRCRAFT CONFIGURATIONS

Flight No.	External Tanks	Lower Vertical Tail	Other	Pilot
2-43-75	on (empty)	Ventral on		Rushworth
2-44-79	off	Ventral on		Rushworth
2-45-81	on	Ventral on		Rushworth
2-46-83	off	Ventral off	Star Tracker	Knight
2-47-84	off	Ventral off	Star Tracker	Knight
2-48-85	off	Ventral off	Star Tracker	Knight
2-49-86	off	Ventral on		Knight
2-50-89	on	Ventral on		Knight
2-51-92	off	Dummy Ramjet	Eyelid	Knight
2-52-96	off	Dummy Ramjet	Full ablative coating, Eyelid	Knight
2-53-97	on	Dummy Ramjet	Full ablative coating, Eyelid	Knight

Flight No. 2-43-75

Early in the planning of the flight program it was decided that the first flight of the aircraft with external tanks would be made with the external tanks empty to allow the pilot to evaluate the aircraft's handling qualities with the external tanks on without the effect of the large cg shifts and the additional complication of operating with a modified propellant system.

One of the first tasks in establishing this flight plan was the selection of a launch point within a gliding distance of one of the existing emergency dry lakes that would allow the aircraft to be flown to the desired tank drop conditions (Mach 2 and $\bar{q}$ = 300 psf) at a point which would allow tank impact to occur in an acceptable area.

The flight was planned and flown from a launch abeam Cuddeback Lake so that tank impact would occur on the Edwards Air Force Base Bombing Range.

The flight was flown using only 50 percent engine thrust in order to achieve the proper ranging from launch to planned tank drop and in addition, closely duplicate the acceleration that would exist with the external tanks full and 100-percent engine thrust. The significant events of the flight plan were as follows:

1. After launch from 45,000 feet at 0.82 Mach number, the pilot was to light the engine and reduce to 50 percent thrust while pulling up to the planned rotation α = 12 degrees. Twelve degrees α was to be maintained until the pitch angle (θ) of 32 degrees was achieved. During the rotation the dynamic pressure was to reach a maximum of 320 psf, well within the desired 400 $\bar{q}$ maximum for tank drop.

2. The planned 32 degrees θ was to occur at 30 seconds and the pilot was to continue to maintain this pitch angle.

3. At 53 seconds from engine light at approximately 58,000 feet the pilot was to push over to and maintain zero g.

4. At 74 seconds (6 seconds before planned tank drop) the pilot was to pull up to 5 degrees α, the desired angle of attack for tank drop.

5. At 80 seconds the aircraft was to be at 2100 fps; 69,000 feet and 300 psf dynamic pressure and the external tanks were to be ejected.

6. After tank separation at 2200 fps, the pilot was to shut down the engine and begin a right turn towards Rogers Lake.

7. The aircraft would then enter the landing pattern on downwind and the remaining propellants would be jettisoned.

8. During the final approach to landing, the pilot was to jettison the lower ventral.

The flight was flown as planned, however, due to an error in the pilot's indicated angle of attack, an average α of 10 degrees was maintained during the rotation and as a result the maximum dynamic pressure was 435 psf. The aircraft's handling qualities were essentially as predicted by the simulator. However, the pilot did comment that the roll stability was less than expected and that the longitudinal control was better than the simulator predicted.

The aircraft trim change at tank separation was 5 degrees nose down as predicted by the simulation. However, the simulation did not include the effect of the inertial reaction of the tanks being ejected from the aircraft. As can be seen in figure 23 this effect resulted in an initial nose up transient of 4 degrees/second pitch rate and a 0.5 g increase in normal acceleration. The maximum transients at tank ejection were:

pitch rate	+5 and -13 degrees/second
roll rate	+20 degrees/second
yaw rate	+2 degrees/second

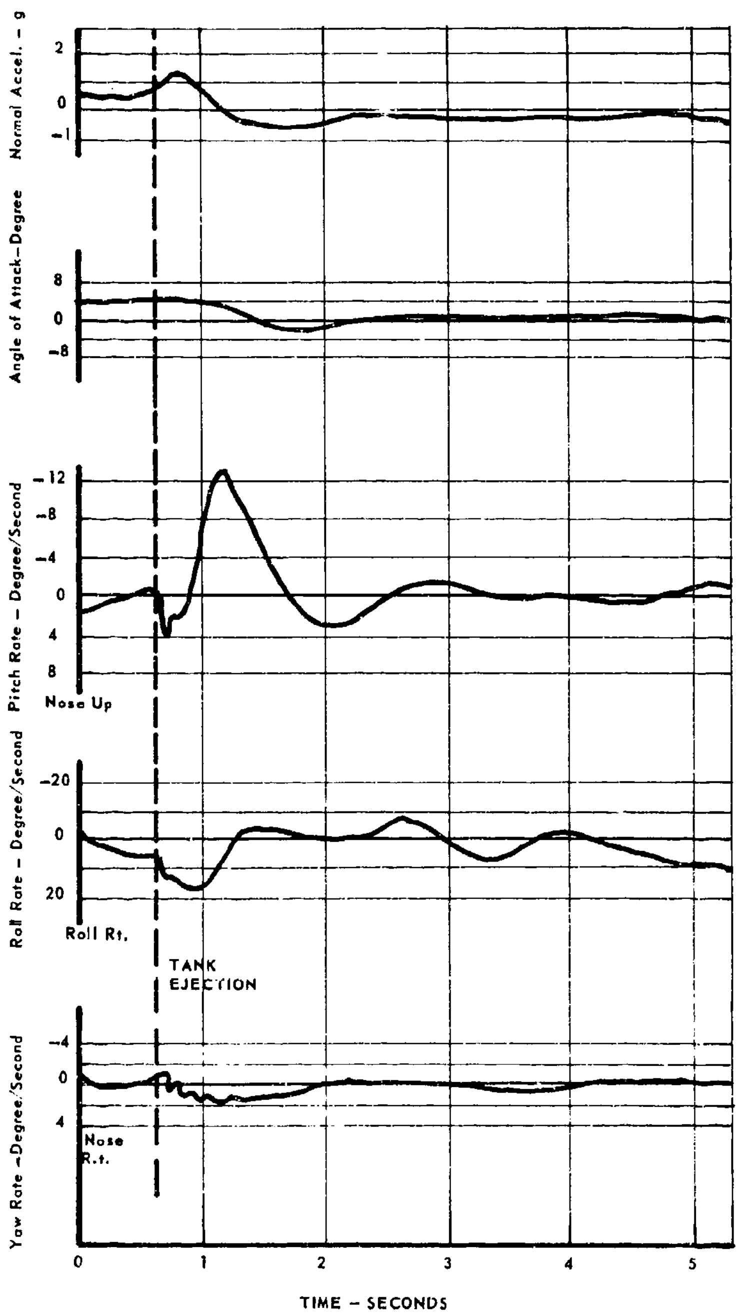


Figure 23. FLT. 2–43 AIRCRAFT TRANSIENTS AT EXTERNAL TANKS EJECTION

However, it should be noted that the weight of the aircraft at tank ejection was less than would exist for a full tank flight because propellants had been consumed from the aircraft's internal tanks.

Immediately after tank drop the pilot shut down the engine and began the space positioning right turn. The maximum velocity was 2220 fps or Mach 2.31. The maximum altitude was 70,600 feet.

Photographic coverage of tank separation was obtained with ground-based cameras with telescopic lenses. Two mobile trackers each with two 150-inch lenses on 35mm Mitchell cameras, were run at 72 and 48 frames per second. A total of six Askania tracking camera were used to record the events of the tank's recovery systems.

Because the resulting image size was so small, only a qualitative analysis of separation could be made. The tanks separated cleanly from the aircraft; however, it did appear that the tanks did not rotate nose down as much as was expected.

The tanks exhibited a tumbling action during flight with the drogue chute attached, and tended to trim at an angle of attack of about -110 degrees. The drogue chutes occasionally collapsed during flight; as a result, the drogue chute riser was lengthened for future flights. The LOX tank nose cone containing the main descent chute did not separate and the tank was destroyed on impact. The ammonia tank was recovered in a repairable condition.

The tumbling action of the tanks increased the total drag over that expected and the tanks therefore fell short of the predicted impact point by approximately 2.5 miles. Both tanks landed well within the bounds of the bombing range.

Flight No. 2-44-79

The main purpose of this flight was to evaluate the aircraft's handling qualities with the lower ventral on at high Mach number at angles of attack up to 8 degrees with the roll and yaw dampers off. The Martin ablative material was applied to the entire left horizontal stabilizer, lower ventral fin and the main landing gear skids. Pressure probes were installed on the leading edge of the jettisonable ventral to determine the local flow conditions in the area proposed for installation of the ramjet.

The planned flight profile was for engine burnout at 100,000 feet with the stability and control evaluation being performed during deceleration between 500 to 700 psf dynamic pressure. The flight was flown on May 18, 1966, essentially as planned, the maximum altitude was 99,000 feet and a maximum velocity of 5430 fps was obtained at engine burnout.

Data analysis of the stability pulses performed indicated that the aircraft had better damping than predicted. However, the pilot's assessment of the handling qualities was degraded by the aircraft being out of lateral-directional trim (possibly because the ablative material was coated only on one stabilizer). The controllability boundary, with the ventral on, was not encountered during the pullup to 6 degrees angle of attack with the roll and yaw dampers off. Quantitative analysis of the heat

protection capability of the ablative material was not possible because of failure of the thermocouple data measuring system. Inspection of the ablative material revealed that it had performed about as expected.

Flight No. 2-45-81

The main purpose of this flight was to evaluate the aircraft's handling qualities with the external tanks full and to expand the envelope of the modified aircraft to 6000 fps at 100,000 feet. Because of military transfer, this was to be the last X-15 flight of the pilot who had participated in the envelope expansion program. In view of this, he was asked to assess the flying qualities to establish whether or not his successor would require a "training flight" with the external tanks empty before advancing to flights with the external tanks full.

As usual for any X-15 flight, alternate procedures were defined for various system failures. For this flight, considerable attention was given to malfunctions of the propellant feed system. Prime concern was the possible loss of control because of extreme cg shifts that could result if the propellant flow from one or both of the tanks failed. Alternate piloting procedures were defined for all of the possible situations conceived. A failure of propellant to flow from either of the external tanks within the first 58 seconds of powered flight, dictated that the mission be discontinued with the pilot shutting down the engine, ejecting the tanks and proceeding to an uprange dry lakebed for emergency landing.

Although a special sensor to provide a positive indication of external propellant transfer was under development, there was no adequate instrumentation in the cockpit on this flight to inform the pilot of the external propellant flow status. A pressure transducer across an orifice in the helium line which pressurized the propellant in the external tanks provided an indirect measure of propellant flow. This parameter was available by telemetry to ground control personnel, and it was established that they would monitor this parameter to advise the pilot of the propellant flow status. Prior to the actual flight, a captive flight (2-C-80) was flown to check the external tank propellant feed system under prelaunch flight environment conditions. The propellant feed system worked as designed, however, the indication of helium flow from the external ammonia tank failed. As a result, the pressure transducer was replaced. It was decided that a pre-launch check of the helium flow during the propellant jettison check four minutes before launch would serve as adequate verification of the helium flow pressure transducer and thereby indicated satisfactory flow from the tanks.

The pre-launch jettison check on flight 2-45 on 1 July 1966 verified helium flow. After launch, however, the ammonia-helium flow did not respond, and approximately 18 seconds after engine light the pilot was advised that "no flow" was observed from the ammonia tank. The pilot immediately began to establish conditions for tank ejection as pre-planned. The aircraft, at the time of the call, was at 12 degrees angle of attack and the pilot pushed over to zero angle of attack and reduced to minimum thrust. At 28 seconds the pilot pulled up to 6 degrees angle of attack and ejected the tanks. Realizing he had forgotten to shutdown the engine, he did so at 33 seconds. A maximum velocity of 1655 fps and a peak altitude of 44,800 feet resulted. The pilot then started a turn to Mud Lake while jettisoning the internal propellants and the emergency

landing was accomplished without incident. After post flight analysis of all data, it was concluded that the ammonia from the external tank was in fact transferring and that the pressure transducer had failed. Therefore the lack of adequate instrumentation had not only resulted in loss of the planned mission, but had also subjected the aircraft to the hazards of a partially full tank ejection and an emergency landing. Although the total mission objectives were not achieved on this flight, the prime objective of evaluating the handling qualities of the aircraft with full external tanks was achieved and it was concluded that the new pilot could fly the aircraft with the tanks full. Figure 24 presents a time history of the first 20 seconds of the flight before the planned mission was aborted.

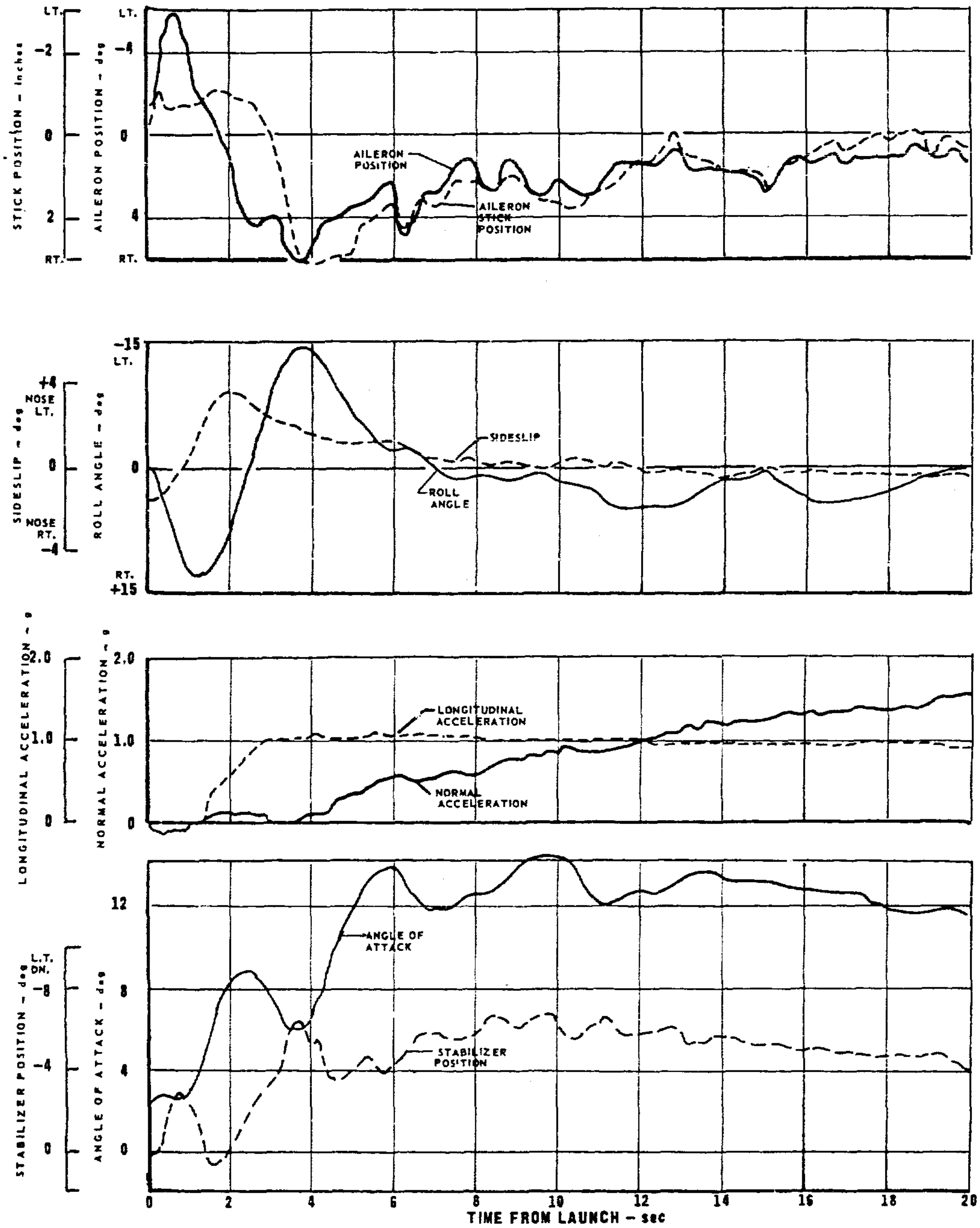


Figure 24 TIME HISTORY OF FLT. 2-45 WITH EXTERNAL TANKS

The launch was described by the pilot as smooth with only 13 degrees of right roll-off occurring. The pilot actuated the throttle as the angle of attack was increasing through 5 degrees and as the thrust built up, the pilot observed the predicted nose down trim change of 3 degrees. The pilot then trimmed to and maintained 12 to 13 degrees angle of attack without difficulty. The roll control task was described as less than that experienced in the simulator; principally due to the roll control forces being higher in the simulator than the aircraft. However, the predicted right aileron required to maintain wings level did exist, as shown in the time history. The effect of the asymmetric cg shift lessening, as propellants were consumed, is quite apparent in the time history as a decrease in aileron required to maintain zero bank angle.

The aircraft's reaction to the ejection of the partially full external tanks was considerably more than that experienced during empty tank ejection. At tank ejection, an initial left roll of 34 degrees/second was induced, followed by a roll to the right of 45 degrees/second. These rates were sufficient to drive the right SAS servo to full deflection for a short period, thereby decreasing the amount of damping available and allowing the oscillation to persist for a longer time period. The tank ejection occurred at 1.61 Mach number at a dynamic pressure of 618 psf. The pilot ejected the tanks with the "FULL" button rather than "PARTIAL" as planned. However, apparently because of faulty circuitry, the tanks ejected in the "partial" mode, i.e., only one ejector cartridge plus the separation rocket. The separation was satisfactory in that the tanks did not recontact the aircraft at the off design conditions of high dynamic pressure and partially full tanks. Attempts to obtain LOX tank accelerations and rotational rates with an FM TM system were unsuccessful. High speed motion pictures of the flight, tank separation, and recovery system operation was obtained from telescopic cameras of the Atomic Energy Commission Tonopah Test Range located near the drop area. The gyrating-tumbling motion of the tanks with the drogue chute was also present on this flight. The recovery system performed satisfactorily, in that, the tanks were recovered in repairable condition. However the mechanism designed to cut the main chute risers on ground contact failed, and the tanks were dragged over the ground by high surface winds.

Flight No. 2-46-83 through 2-48-85 (Ultra Violet Stellar Photography Experiment)

These three flights of the aircraft were flown to accomplish the Ultra Violet Stellar Photography experiment. Star orientation at this time of the year, once again made it possible to pursue the experiment. The purpose of the experiment was to obtain measurements of stellar brightness in the spectral region between 1800 and 3200 Angstroms, which cannot be observed from the ground because of the ozone layer. The experiment was successful and the results are documented in reference 5.

Flight No. 2-49-86

The prime purpose of Flight 2-49-86, flown on 30 August 1966, was to allow a different pilot to become familiar with the aircraft's handling qualities with the lower ventral on prior to the next flight which would be with both the external tanks and the lower ventral. The flight was flown as planned, and was essentially the same as Flight 2-44-79. A maxi-

mum velocity of 5190 fps was achieved at engine burnout. Dampers off stability data were obtained during the deceleration and the results correlated very well with those from Flight 2-44-79.

The Maurer camera was carried in the 29-inch extension bay as part of the Induced Turbulence Experiment to study the effects of hypersonic flight environments on high resolution photography. Portable resolution targets were positioned on the ground along the planned track. This made it necessary for the pilot to fly a precise heading in order to pass over the target. This was achieved by using a precision heading indicator.

Flight No. 2-50-89

This first successful full tank flight was a major milestone in the envelope expansion program. The maximum velocity attained was 6250 fps (6.33 Mach Number) at an altitude of 96,800 feet. The maximum altitude during the flight was 98,900 feet.

The improved sensor for indicating external propellant flow was installed in the aircraft for this flight. A small hinged plate was installed into each external propellant transfer line. External propellant flow into the aircraft's tanks deflected these "paddle switches" completing an electrical circuit which then illuminated two green lights on the cockpit panel indicating flow from each external tank. This system was exercised during an engine ground run and indicated flow at even the low flow rates associated with the pre-launch checks of jettison and engine second stage operation. This system was to be used in flight as the prime indication of satisfactory propellant transfer.

The aircraft was launched from the East Mud Lake launch point, 185 NM from Edwards. Again the launch was smooth with only 12 degrees of left roll-off occurring: however, the pilot made too large a nose up input causing the aircraft to momentarily overshoot the planned 12 degrees angle of attack by 5 degrees. The increased lift resulting from the higher angle of attack caused a higher rolling moment about the laterally offset cg. As a result the aircraft rolled off 28 degrees to the left, requiring considerable pilot attention and control input to level the aircraft. During the rotation the pilot maintained an average of 12 to 13 degrees indicated angle of attack; however, the actual angle of attack was 2 degrees less due to a calibration shift. Therefore, the maximum dynamic pressure during rotation was 730 psf rather than the expected 600 psf. The planned pitch angle of 35 degrees was maintained within ± 1 degree. The pilot ratings for the specified control tasks are shown in table III. No pilot rating was given for the yaw axis because no control inputs were required.

Table III

PILOT RATINGS FOR CONTROL TASKS
WITH EXTERNAL TANKS (FLIGHT 2-50-89)
(Based on a Pilot Rating Scale of 1 to 10)

Task	Pitch Rating	Roll Rating
Acquiring angle of attack	3	3.5
Maintaining angle of attack	2.5	2.5
Acquiring pitch angle	2	2
Maintaining pitch angle	2	2

TABLE IV

TANK EJECTION CONDITIONS (FLIGHT 2-50-89)

	Planned	Actual
Velocity, fps	2100	2170
Mach No.	2.16	2.27
Altitude, ft.	69,000	69,500
Dynamic Pressure, psf	320	342
Angle of Attack, deg	5	3.5

As planned the pilot pushed over to an indicated angle of attack of 5 degrees at 59 seconds to set up for tank ejection. The pilot switched to "internal" flow at 64 seconds which closed the valves in the external-internal transfer lines. At 67.5 seconds, the external tanks were ejected. The transients induced to the aircraft were similar to those experienced on the empty tank flight (2-43-75). The pilot was impressed with the "bang" and "jolt" when the ejection occurred. Tank ejection conditions are listed in table IV.

After tank drop the aircraft continued to climb and accelerate at approximately zero angle of attack. At 4500 fps the pilot extended the speed brakes and increased the angle of attack to maintain constant altitude. The pilot shut down the engine as planned at 6000 fps on the inertial velocity indicator; however, final radar data proved the maximum velocity to be 6250 fps. Engine burn time was 136.2 seconds compared to the planned 132 seconds. The deceleration was accomplished with the speed brakes extended and thus the time at high Mach numbers was minimized. The time from maximum velocity to Mach 3 was 138 seconds. This rapid deceleration kept the aircraft temperatures lower than would have been experienced with a gradual deceleration. However, the maximum temperatures recorded were higher than experienced during previous flights.

The canopy glass test discussed previously and shown in figure 20 was flown on this flight.

The external tank recovery system operation was satisfactory despite the fact that the "manhole cover" drain valves, which allow the unusable propellants to be expelled, failed to open. The tanks impacted within the restricted area at elevations of 5700 and 6700 feet. Recovery was performed at a later date by helicopter. Detailed analysis of actual tank trajectories were not made. Radar data of the ammonia tank prior to main chute deployment indicated the tank traveled 2 miles from the release point. The idealized no wind calculations had predicted a distance of 6 miles. This difference is attributed to the increased drag associated with the tumbling tank.

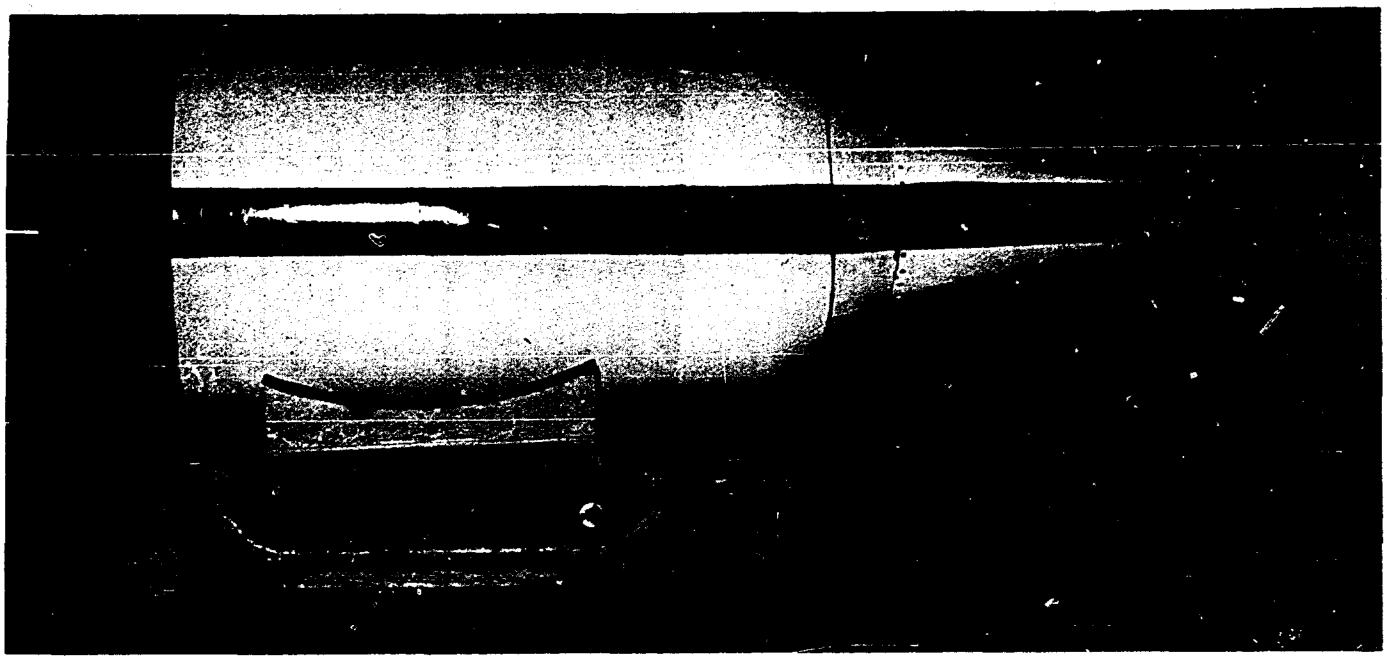


Figure 25 DUMMY RAMJET

● AIRCRAFT CONFIGURATION CHANGES

After the successful demonstration of the advanced design on Flight 2-50-89, the envelope expansion program was reoriented. In order to preclude the requirement for separate envelope expansion programs for different aircraft configurations, it was decided to reconfigure the aircraft to its final aerodynamic configuration.

During the following winter rainy season when the X-15 could not fly because of water on the lake beds, preparations were made for this new phase of the program. A "dummy" ramjet shape was designed to be mounted in place of the lower movable ventral (figure 25). A parachute recovery system was included in the design to allow the unit to be recovered after jettison on the landing approach. Since some refurbishment would be required after flight, three of the dummy ramjets were constructed. Forty-two inches of the forward part of the fixed ventral were cut off and the remaining portion of the ventral configured as a ramjet pylon. Other configuration changes were the installation of the canopy eyelid, installation of Yaw ASAS, and the removal of the ballistic control system rockets.

Limited wind tunnel data were obtained on the basic ramjet configuration without external tanks at Mach numbers of 1.5, 3.0, and 6.5. Incremental effects of replacing the existing movable ventral with the ramjet were determined from the wind tunnel tests and this increment was applied to the ventral off data in the simulator mechanization. The increment at 1.5 Mach number was assumed to apply at all lower Mach numbers in the simulation. Wind tunnel data with the tanks on were obtained at only 1.5 Mach number. The stability derivatives for this configuration were mechanized by adding the incremental differences to the existing ventral on derivatives based on the increment at 1.5 Mach number. Since the ramjet is fixed, the rudder effectiveness derivatives ($C_{n_{\delta r}}$,

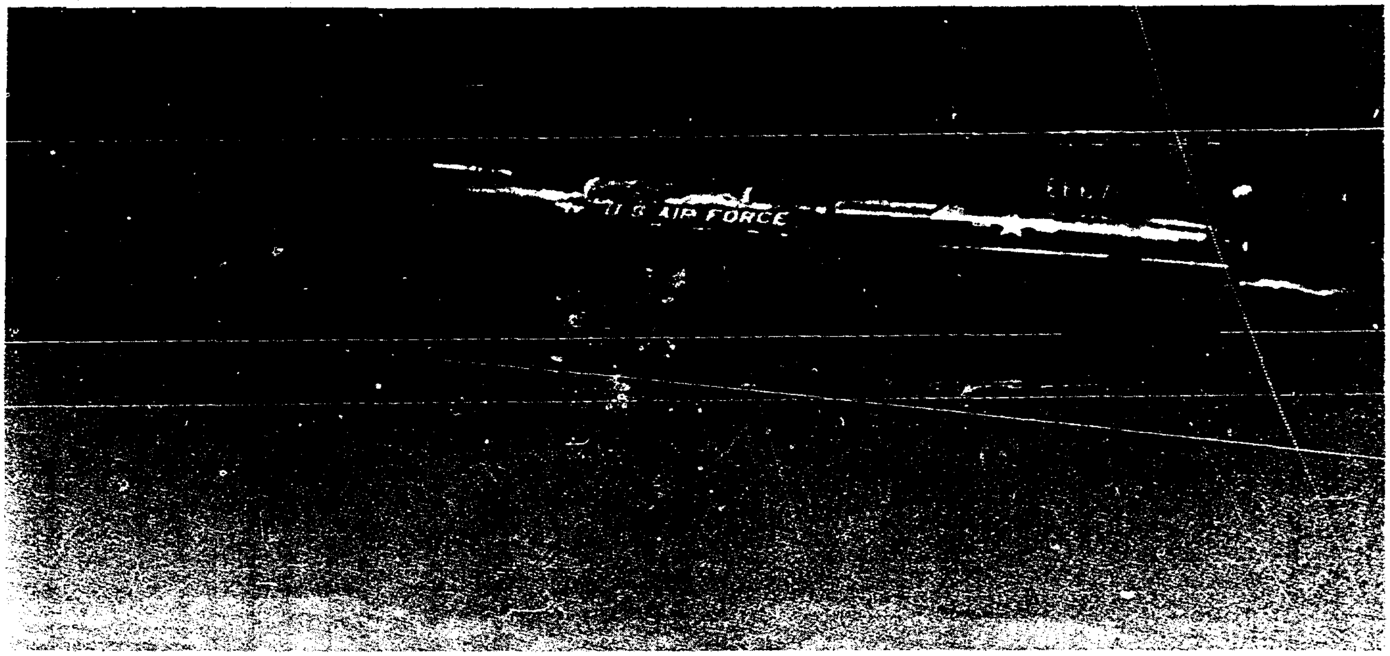


Figure 26 INFLIGHT PHOTOGRAPHY FLIGHT No. 2-51-92

Cl_{δ_r}) with the movable ventral were not applicable; therefore the ventral off rudder effectiveness were assumed valid. The lack of wind tunnel data below Mach 1.5 resulted in some degree of uncertainty in simulator validity at transonic and subsonic speeds. However the handling qualities determined on the simulator were satisfactory even when the derivatives were degraded by 30 percent. The predicted lateral-directional handling qualities were essentially the same as with the ventral on. However, the longitudinal trim characteristics were predicted to be quite different, particularly at low angles of attack. The additional frontal area of the ramjet located below the cg resulted in a nose down pitching moment that required additional nose up stabilizer. This predicted longitudinal trim difference was particularly apparent on the simulator in that frequent nose up trimming was required to maintain zero normal acceleration as the Mach number increased.

Flight No. 2-51-92

Evaluation of the aircraft's handling qualities with the dummy ramjet engine installed was the main purpose of this flight flown on 5 May 1967 (figure 26). In planning the flight it became necessary to consider a new constraint. The canopy eyelid was designed to be used when the aircraft was coated with ablative material and thus to keep the temperatures below the design level, the eyelid itself had to be covered with ablative material. This presented the problem that undesirable thermal stresses could result from temperature differentials between the cooler structure behind the ablative coated eyelid and the unprotected structure on the canopy below the eyelid. The real-time temperature simulation was used to establish the flight plan which best satisfied the objectives of the flight within the constraint imposed by the canopy temperature limitation. The resulting flight plan limited the maximum velocity to

4500 fps and thus dictated a launch closer to Edwards than Mud Lake. The flight was flown from the Hidden Hills launch point which is 121 NM from Edwards.

The flight was flown essentially as planned, 10 degrees angle of attack rotation, 25 degrees pitch angle climb, pushover to zero g at 46 seconds, and shutdown at 4500 fps. The actual maximum velocity achieved was 4750 fps due to an error in the indicated velocity.

As predicted, continuous noseup trimming was required by the pilot to maintain zero g.

A series of dampers off maneuvers were performed during the deceleration. The lateral-directional handling qualities were acceptable and even better than those experienced on the simulator up to the maximum angle of attack (8 degrees).

During the approach to high key, the pilot opened the canopy eyelid at approximately 2.2 Mach number. This opening was accompanied by a slight trim change inn all three axis (nose up, roll right, right yaw). An abrupt nose up longitudinal trim change occurred when the aircraft was decelerating in the transonic range. This was unexpected since no wind tunnel data had been obtained below Mach 1.5: however, the pilot was able to trim nose down at a sufficient rate to counteract the trim change.

On final approach the pilot jettisoned the ramjet, and there were no aircraft trim changes associated with the event. Ramjet separation characteristics were calculated by the contractor and under certain conditions the ramjet could recontact the aircraft, therefore a recommended ejection envelope and cg limits for the unit were established. To obtain data on the actual ramjet separation, a mobile tracker with telescopic cameras was located normal to the final approach track at the point of expected ramjet ejection. The pictures obtained were excellent and detailed analysis of the film was possible (figure 27). The agreement of the actual separation with the theoretical calculations was good, with the actual clearance being better than predicted (figure 28). The ramjet recovery system operation was unsatisfactory. The stranded cable attaching the recovery parachute to the ramjet failed as the chute deployed: however the impact damage was not major and the unit was refurbishable.

Flight No. 2-52-96

After Flight 2-51-92 on May 8, 1967, preparations were begun to configure the aircraft for the first flight with a complete ablative coating. The application of the ablative coating was begun on 25 May and was completed in five weeks requiring approximately 2000 man-hours (figures 29, 30, 31, and 32). The design of the premolded gloves covering the leading edge of the wings, tail surfaces, and canopy was based on a Mach 8 design mission. The application of the sprayed on ablative material was based on the expected heating rates on a more realistic maximum design mission to 7.4 Mach number and the established requirement to limit the undersurface temperature to 600 degrees F because of loss of bond strength at higher temperatures. The thickness varied from 0.65 inches at the leading edge of the horizontal stabilizer to 0.02 inches at locations on the upper surface of the wings.

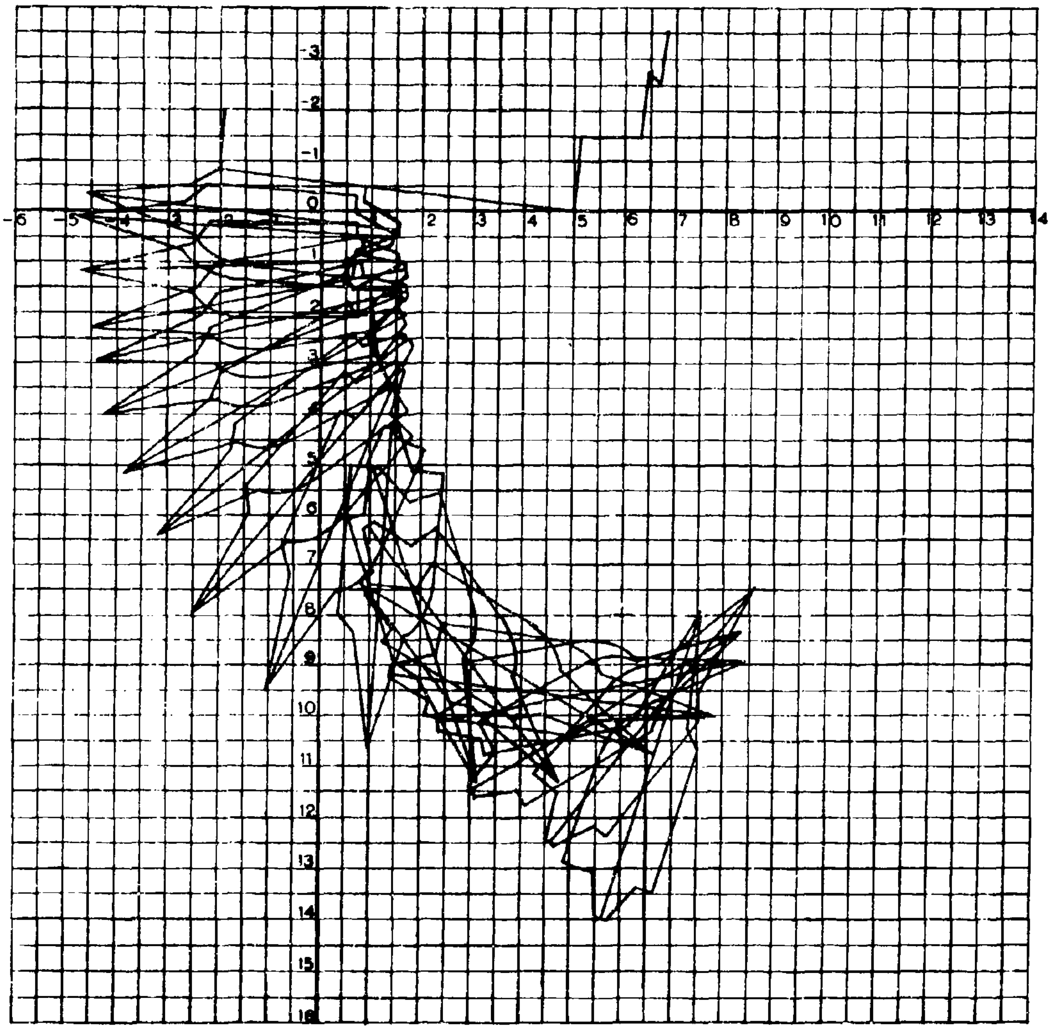


Figure 27 FLIGHT 2-51-92 DUMMY RAMJET SEPERATION FROM MOBILE TRACKER PICTURES

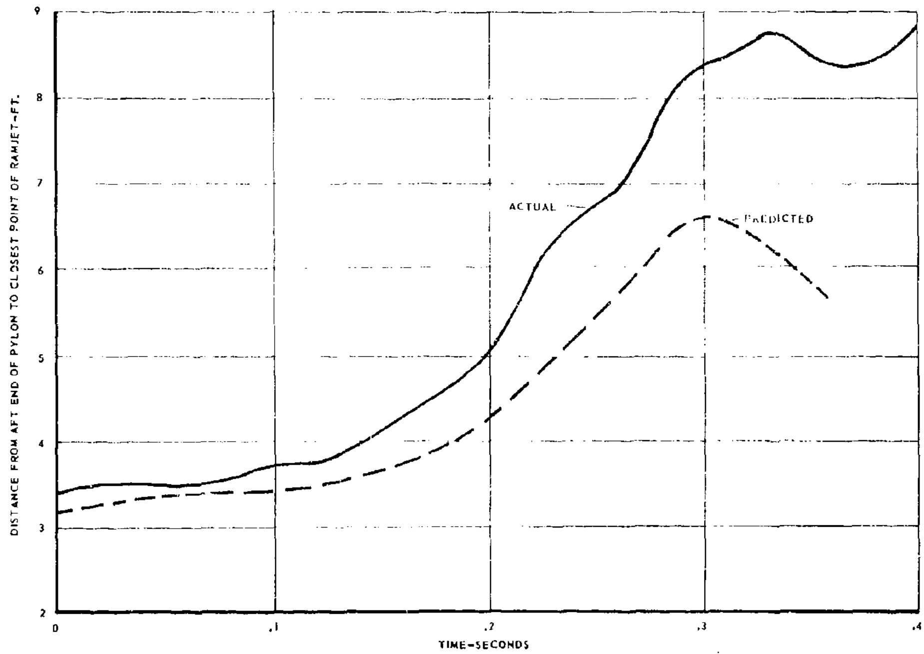


Figure 28 COMPARISON OF ACTUAL AND PREDICTED RAMJET SEPARATION FLT. 2-51

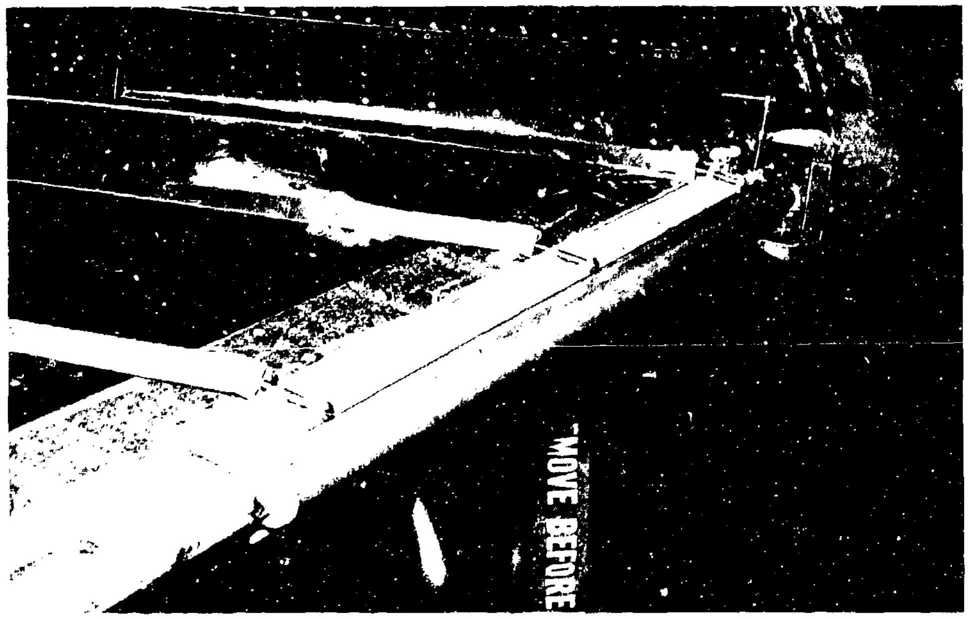


Figure 29 APPLICATION OF PRE-MOLDED ABLATIVE MATERIAL ON WING LEADING EDGE

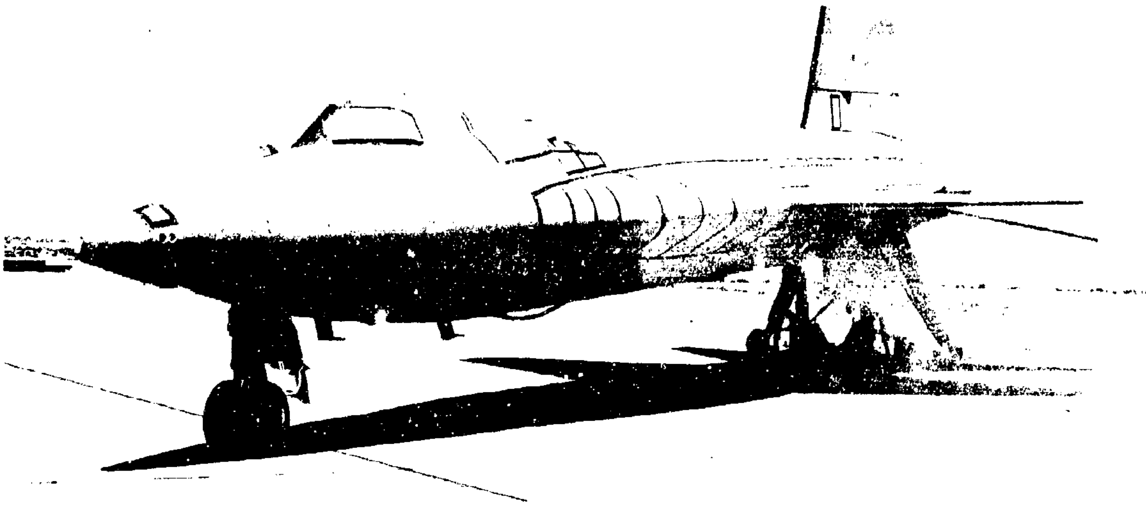


Figure 30 THREE–QUARTER FRONT VIEW OF AIRCRAFT WITH ABLATIVE COATING–WITH ACCESS PANELS EXPOSED

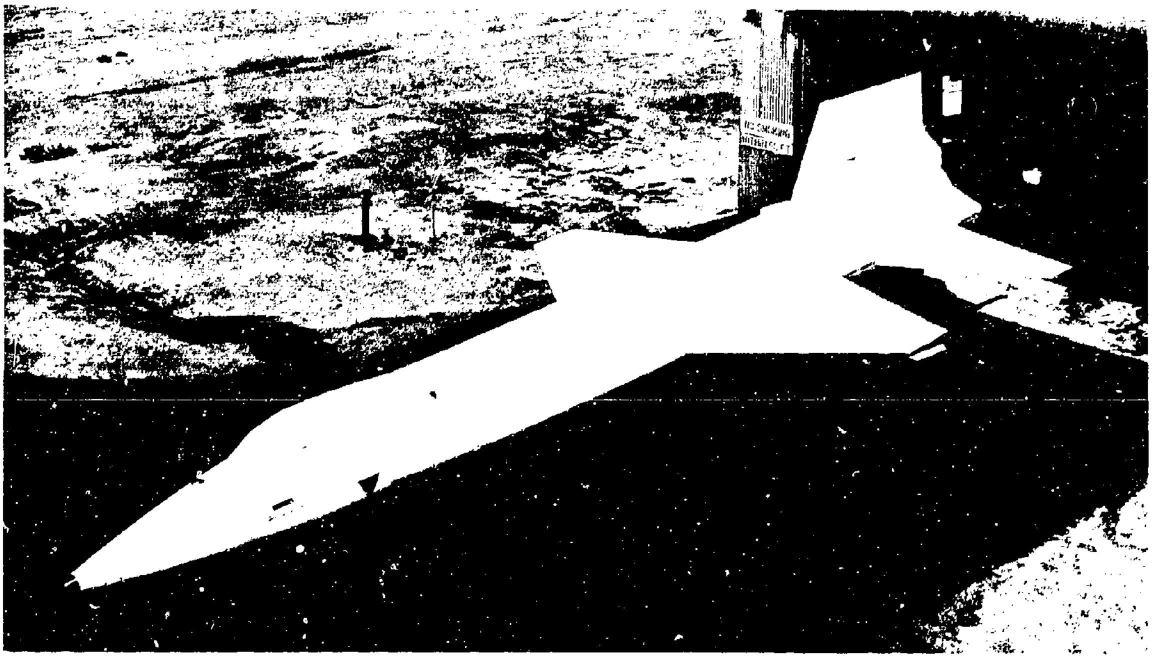


Figure 31 THREE–QUARTER FRONT VIEW OF AIRCRAFT WITH ABLATIVE COATING PROTECTED WITH WHITE SEALANT

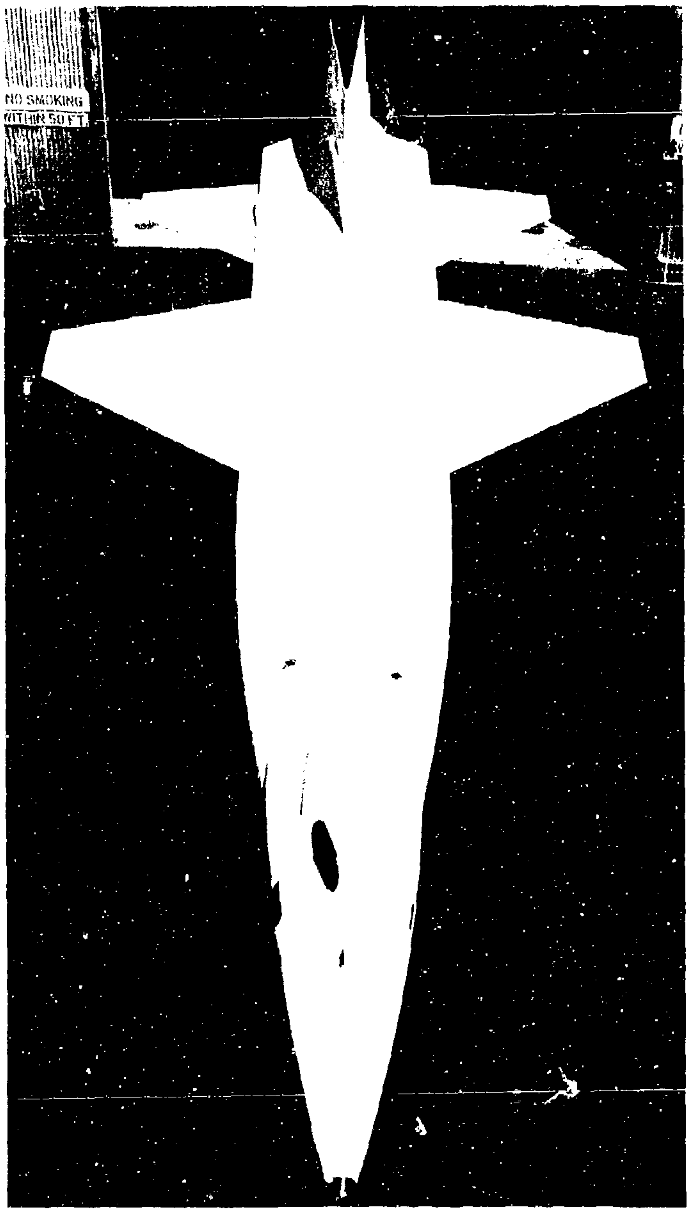


Figure 32 TOP–FRONT VIEW OF AIRCRAFT WITH ABLATIVE COATING PROTECTED WITH WHITE SEALANT

One of the uncertainties of the flight plan was the amount of performance degradation caused by the ablative material. The increase in the drag coefficient caused by the increased leading edge radius, trailing edge thickness and skin friction was theoretically estimated to be 0.015. This increment was introduced into the simulator as part of the data on which the flight plan was based. During pilot preparation on the simulator, this parameter was varied to acquaint the pilot with possible deviations from the planned profile. To acquaint himself with possible changes in energy management at landing pattern speeds due to a reduction in L/D, the pilot practiced the X-15 approach in the F-104 at less than normal X-15 L/D.

In considering the results of possible system failures, it was recognized that a failure of the pilot's attitude indication at high speed could possibly leave the pilot without adequate roll reference. Outside reference would not be available because the left window would be covered with the eyelid, and the aircraft flight conditions would be outside the limits for opening the eyelid. Also, the right window could become coated with ablative residue. Therefore an F-104 attitude system was installed which included a two-inch indicator in the cockpit panel. Figures 33, 34 and 35 show the cockpit configuration on the last two flights of the aircraft (2-52-96, 2-53-97).

Prior to the flight, a planned captive flight was made to check the aircraft systems after coldsoak at altitude. This check was deemed desirable to determine if the presence of the ablative coating on the aircraft changed the environment inside the aircraft enough to affect the aircraft systems. The external tanks were installed on the aircraft for this captive flight. Although some aircraft discrepancies were discovered and later corrected they were not attributed to the environment. The external tanks were removed prior to flight 2-52-96.

Flight 2-52-96 was flown on 21 August 1967. A maximum speed of 4939 fps was obtained at engine burnout. The boost profile from Hidden Hills was flown essentially as planned. After burnout the pilot performed a series of stability and control maneuvers. No differences in handling qualities were detected that could be attributed to the presence of the ablative material.

While approaching the pattern, the pilot actuated the alternate pitot tube. After the aircraft had slowed to a subsonic speed, the pilot compared the indicated airspeeds from the alternate pitot-static sources with the standard system and airspeeds called from the chase aircraft. The airspeeds from the alternate sources were 50 to 70 knots higher than those from the standard sources, but showed the closest agreement with the chase aircraft; therefore, the airspeed from the alternate system was used for landing. On the previous flight without the ablative coating, the difference between the alternate and standard airspeeds had been noted to be only 20 to 30 knots.

The performance of the ablative material on this relatively low speed flight was very good (figure 36). Most of the degradation occurred on the leading edges while remaining areas showed little change in appearance. Two localized problems occurred during the flight. Small 1.5-inch sections of the ablative material separated from the left side of the upper vertical stabilizer. This separation occurred at the interface of

two separate spray coatings (figure 37). The failure was attributed to the spray mixture being too dry at the start of the second coat. A tensile test of the ablative coat in this area prior to the flight had indicated that the adherence was slightly sub-marginal, but was not considered critical for this flight. The second problem was erosion of the ablative material on the leading edge of the ramjet pylon (figure 38). It was believed that this increased erosion was the result of shock wave impingement from the ramjet, however the seriousness of the problem was not fully appreciated until the next flight.

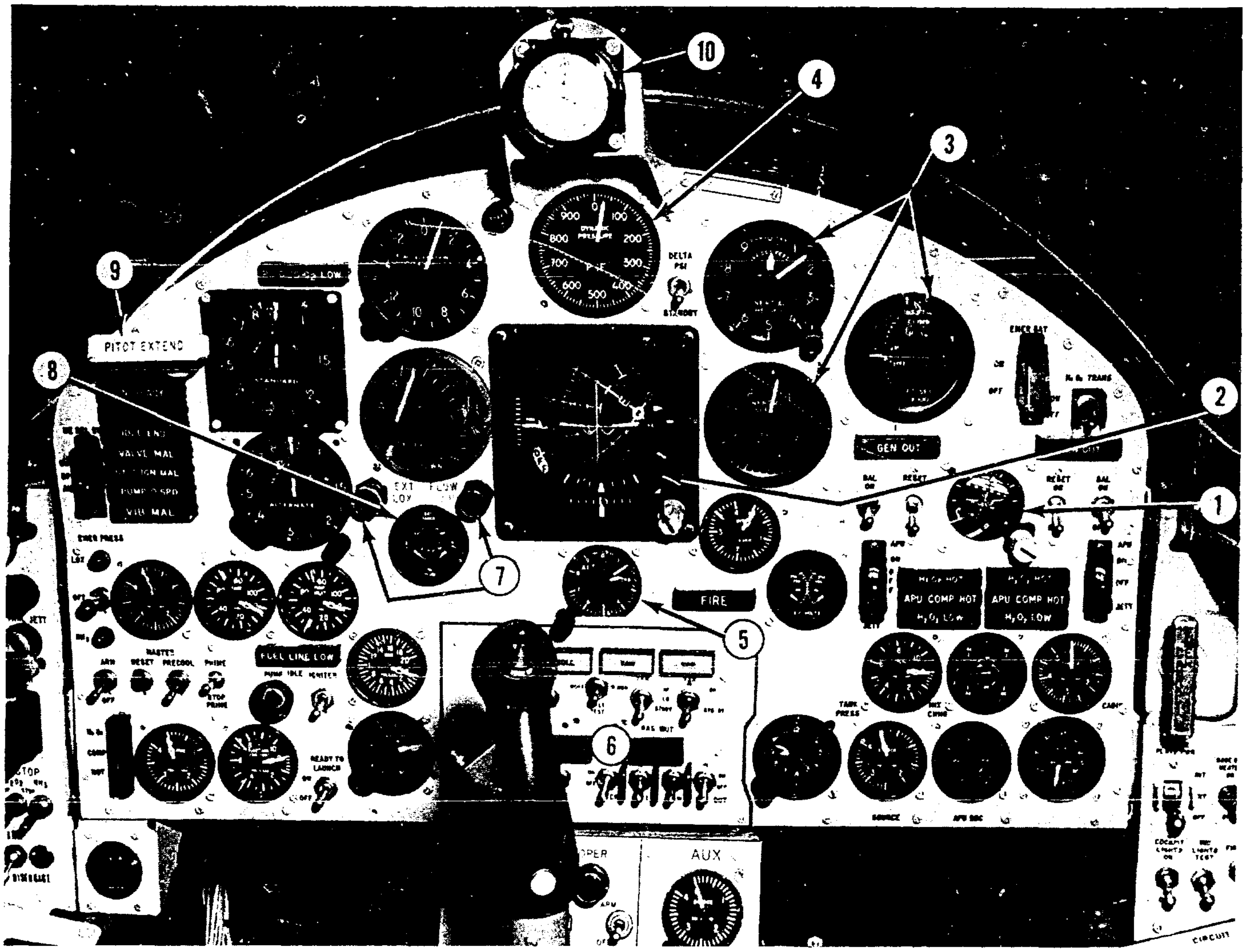


Figure 33 COCKPIT PANEL

1. ALTERNATE ATTITUDE INDICATOR (2 AXIS)
2. PRIMARY ATTITUDE INDICATOR (3 AXIS)
3. INERTIAL INDICATORS (VELOCITY, ALTITUDE, RATE OF CLIMB)
4. DYNAMIC PRESSURE (FROM BALL NOSE)
5. PRESSURE ALTITUDE FROM ALTERNATE STATIC SOURCE
6. SAS CONTROL PANEL
7. EXTERNAL PROPELLANT FLOW LIGHTS (FROM "PADDLE SWITCHES")
8. EXTERNAL PROPELLANT FLOW INDICATOR (FROM HELIUM PRESSURE)
9. ALTERNATE PITOT TUBE EXTEND LEVER
10. ENGINE BURN TIME

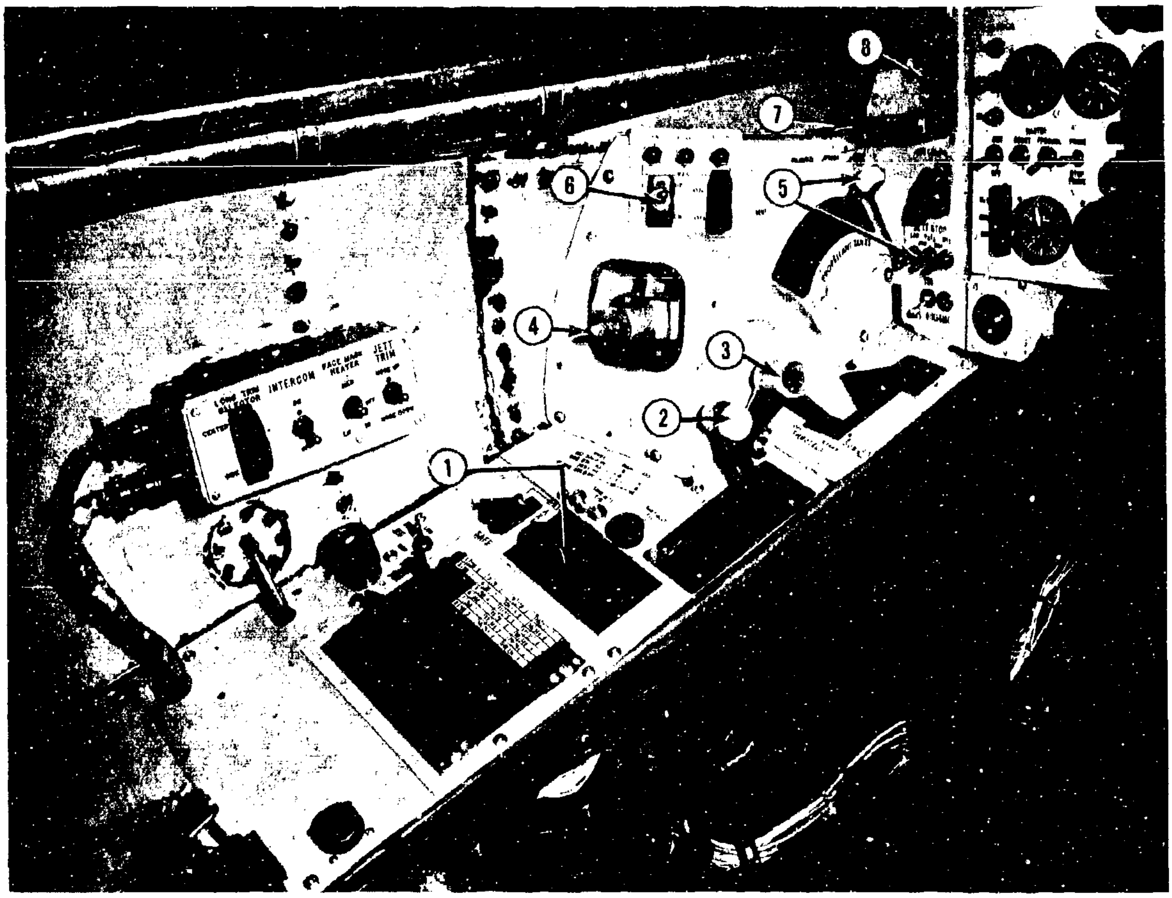


Figure 34 LEFT COCKPIT CONSOLE

1. EXTERNAL TANK CONTROL PANEL
2. SPEED BRAKE HANDLE
3. THROTTLE
4. BCS HANDLE REMOVED
5. PROPELLANT JETTISON CONTROLS
6. LANDING FLAP SWITCH
7. LANDING GEAR HANDLE
8. VENTRAL OR RAMJET JETTISON SWITCH

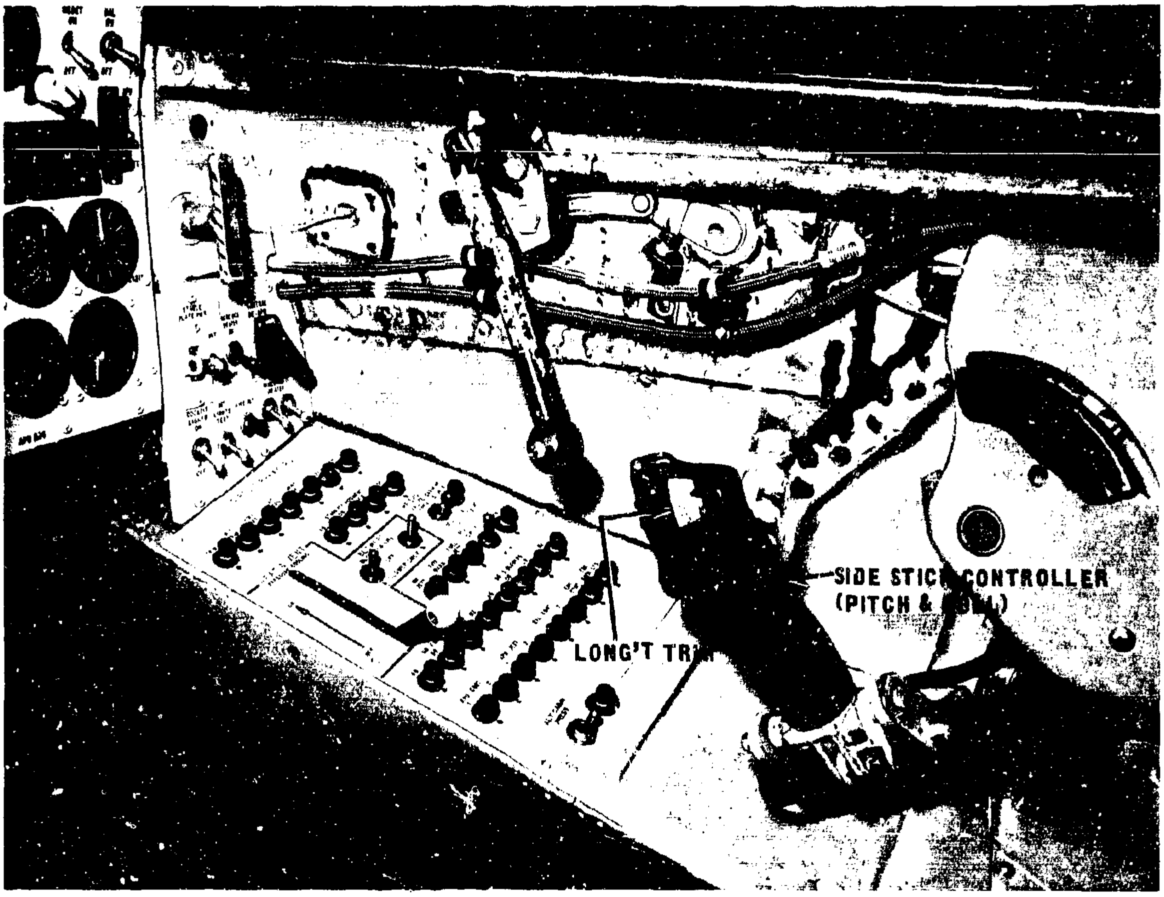


Figure 35 RIGHT COCKPIT CONSOLE

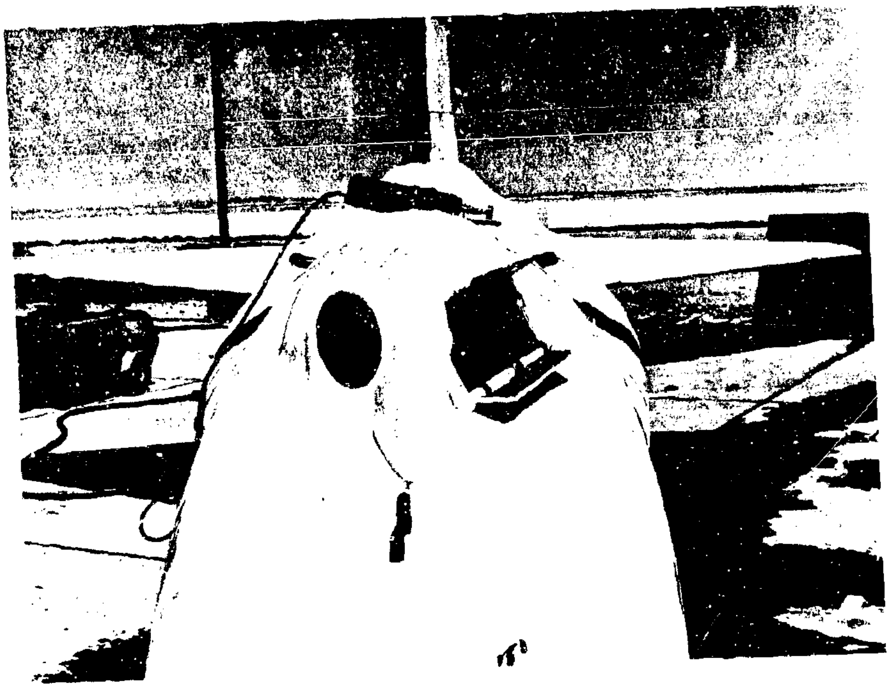


Figure 36 FRONT VIEW OF AIRCRAFT WITH ABLATIVE COATING AFTER FLIGHT No. 2-52-96

Figure 37 VERTICAL TAIL AFTER FLIGHT No. 2-52-96

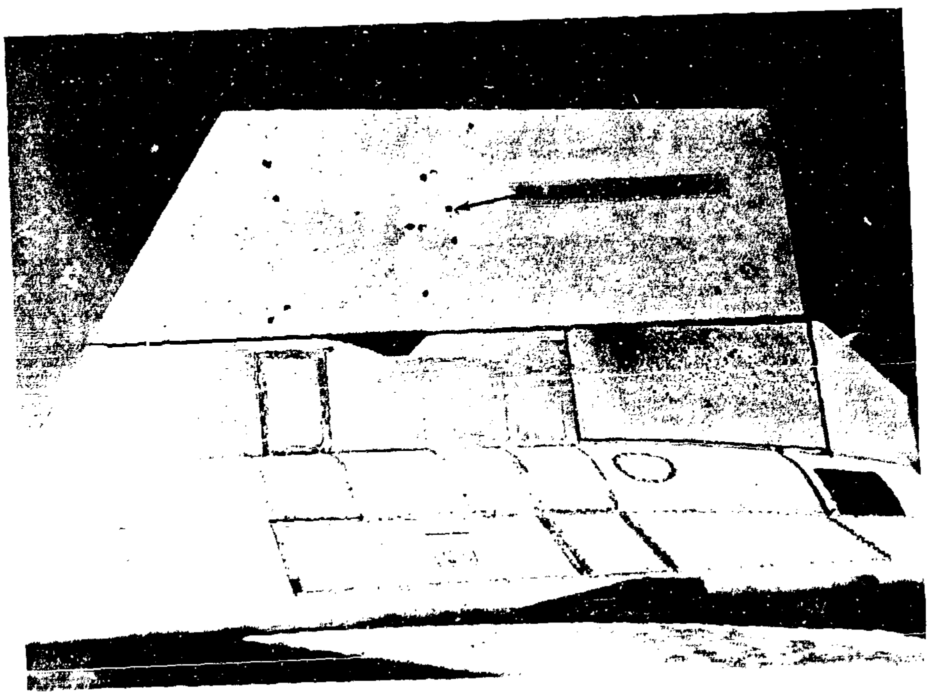


Figure 38 RAMJET PYLON LEADING EDGE AFTER FLIGHT No. 2-52-96

Flight No. 2–53–97

With the successful demonstration of satisfactory handling qualities with the ablative coating on the aircraft and effectiveness of the ablative material as a heat shield, the aircraft was cleared for the Mach 6.5 envelope expansion flight.

Aircraft preparation for this flight consisted mainly of refurbishing the ablative material wear resulting from the last flight. This task required approximately 700 man-hours over a two week period.

The published flight plan is presented in appendix I. The first page lists the key piloting events associated with trajectory control to achieve desired conditions for tank ejection and then to attain 100,000 feet. The planned maneuvers after shutdown were to verify the stability and control derivatives and to establish longitudinal trim characteristics. Extension of the speed brakes after shutdown was for energy management purposes. Note that an allowance for the increased drag due to the ablative material was included in the simulation. Initially a drag coefficient of 0.015 was applied; once the velocity increased to 5500 fps, it was assumed that the ablative material would increase in roughness therefore the drag increment was changed to 0.02 and remained at this level for the rest of the simulated flight.

During mated flight while outbound to the launch point, the systems were operationally checked prior to committing the X-15 to launch. A partial list of those items which are subject to change because of the type of flight or are new items are listed on the flight plan. The first four listed on this flight plan were standard for X-15 operations. Items 5, 6, and 7 were peculiar to this flight because of the external tanks and ablative material. One of the unlisted requirements for an X-15 launch was that all of the support aircraft and ground equipment be at the proper location. To illustrate the amount of support associated with a flight, figure 39 depicts the support used for this flight.

In the interest of flight safety the major aircraft systems had to function normally for the flight to proceed to the planned speed. Malfunctions possible during the flight dictated that a preplanned alternate flight be flown. As seen in the flight plan, the maximum speed was limited to 5400 fps in case of failure of any of the dampers, attitude indication, ball nose (α, β, $\overline{q}$) and/or external propellant flow failure at certain times.

The ground rule for a failure of the engine to light on the first attempt after launch on previous tank flights had been to immediately eject the external tanks and fly an alternate profile. The main concern was that the maximum allowable dynamic pressure of 1000 psf would be exceeded during the rotation (due to altitude loss) and that the dynamic pressure at the planned tank ejection would be too high for good separation characteristics. The possibility that a good mission could be lost to this cause was very real because delayed engine lights had occurred on six X-15 flights. For this flight the ground rule was changed to allow one restart attempt. The change was based in the following factors:

1. Increased confidence from previous flights in handling qualities at the planned α with the tanks on.

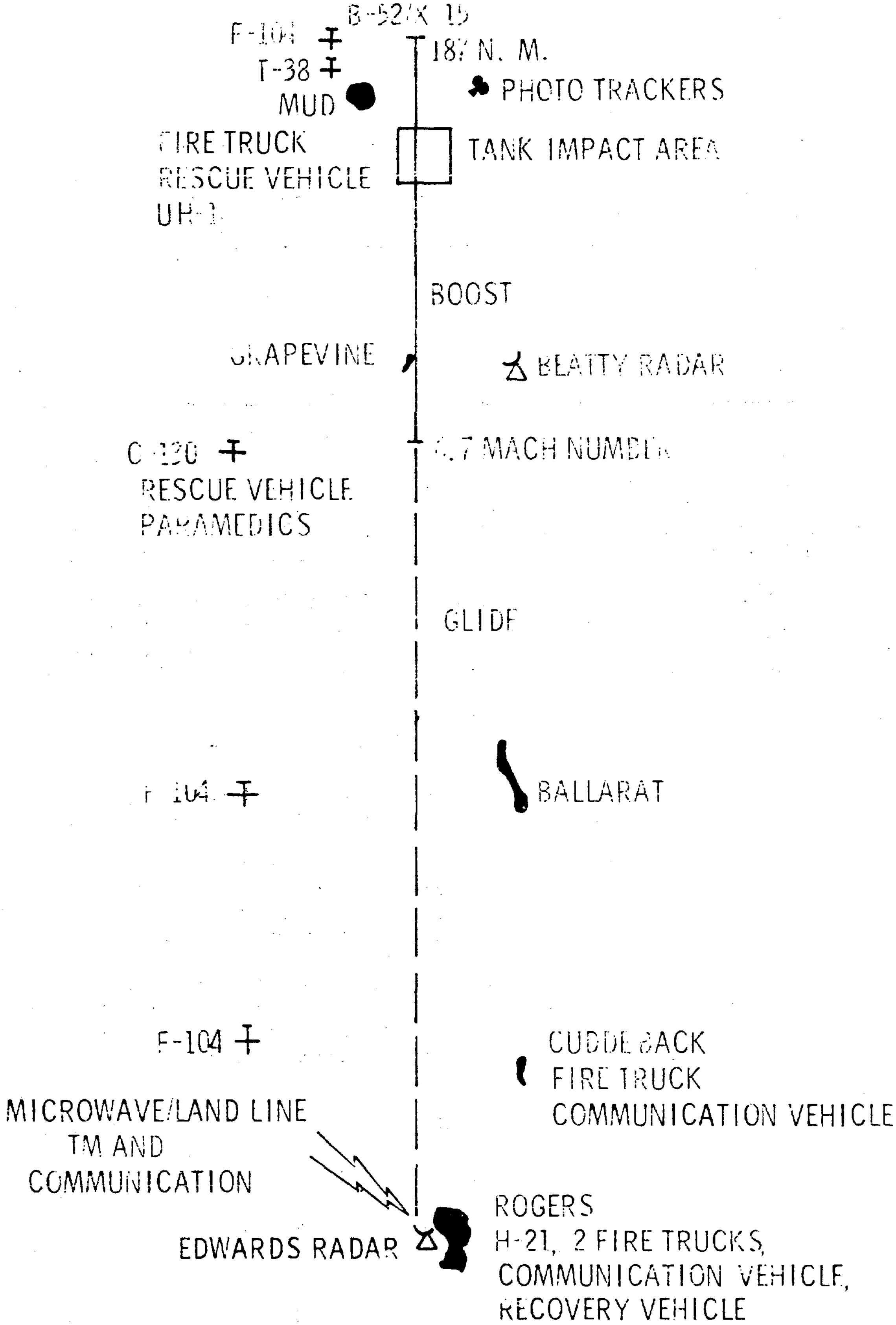


Figure 39 X–15 A–2 FLIGHT 2–53–97 OPERATION SUPPORT

TABLE V

TANK EJECTION CONDITIONS

FLIGHT NO. 2-53-97

Velocity	2170 fps
Altitude	73,500 feet
Dynamic pressure	288 psf
Mach number	2.2
Angle of attack	4.2 Degrees

2. Simulator studies showed that the rotation could be performed at less than 1000 psf dynamic pressure, particularly if the pilot flew at 2 degrees higher α and limited normal acceleration to 2.4 g. In addition, reduction of the throttle could be used as a positive method to keep the dynamic pressure less than 1000 psf if necessary.

3. Simulator studies also indicated that if the dynamic pressure was too high at the time of planned tank ejection, no detrimental effects resulted if the ejection was delayed until the dynamic pressure (which would be decreasing at this time) reached the desired value.

By flight date all the various alternate procedures were well known to the pilot after having practiced on the simulator for 35 hours. Fortunately, it did not become necessary to utilize an alternate profile since the flight proceeded basically according to plan.

The launch transients were very mild with a bank angle excursion of 14 degrees. During the rotation the pilot had good control of the aircraft and increased the angle of attack to 15 degrees and felt the onset of buffet. The remainder of the rotation to the planned pitch angle was made at 12 to 13 degrees angle of attack. During this period the roll control was excellent and the bank angle did not deviate more than 8 degrees. The maximum dynamic pressure experienced during the rotation was 560 psf, close to the 540 psf observed on the simulator. The planned pitch angle of 35 degrees was reached in 38 seconds and was maintained within plus/minus one degree.

The external tanks were ejected 67.4 seconds after launch at the conditions shown in table V. Tank separation was satisfactory, however, the pilot felt the ejection was "harder" than the last one he had experienced (Flight No. 2-50-89). The longitudinal trim change to the aircraft was from 4.2 to -2 degrees angle of attack. The external tank recovery system performed satisfactorily and the tanks were recovered in repairable condition.

After tank ejection the planned 2 degree angle of attack was maintained within ± 1 degree. As the aircraft came level at an indicated alti-

tude of 99,000 feet, the pilot increased the angle of attack to 6 degrees to maintain zero rate of climb. During this task the pilot reported that the pitch control was very sensitive and it was difficult to hold a constant angle of attack.

The pilot reported shutting down the engine at 6500 fps; however, the final radar data analysis revealed the maximum velocity to be 6630 fps. The total engine burn time was 141.4 seconds, which compared favorably with the 141 seconds planned. However, the aircraft had achieved a velocity which was 130 fps faster than that of the simulator during this time.

During the deceleration the pilot was concentrating on performing stability and control maneuvers and as a result the profile was not exactly as planned. Figure 40 presents a comparison between the planned and actual profiles. Note that after shutdown the aircraft did not descend at the rate planned, resulting in a lower dynamic pressure between 5500 and 4000 fps. This anomaly, along with the higher maximum velocity, presented the pilot with the task of managing higher energy in approaching the high key position. Figure 41 presents a comparison between the planned and actual velocity as a function of distance from high key. The region of largest dispersion from the planned ranging occurred at the time when the dynamic pressure was lower than planned. To regain the desired high key energy conditions, the pilot delayed the retraction of the speed brakes and flew the remainder of the deceleration at a higher dynamic pressure (a maneuver commonly used on X-15 flights).

The ability of the ablative material to protect the aircraft structure from the high aerodynamic heating was considered good except in the area of the dummy ramjet where the heating rates were significantly higher than predicted. Considerable heat damage occurred on the dummy ramjet and the ramjet pylon. The ramjet instrumentation ceased approximately 25 seconds after engine shutdown indicating that a burn through of the ramjet/pylon structure had occurred. Shortly thereafter the heat propagated upward into the lower aft fuselage area causing the engine hydrogen peroxide hot light to illuminate in the cockpit. Ground control, assuming a genuine overheat condition, requested the pilot to jettison the remaining engine peroxide. The high heat in the aft fuselage area also caused a failure of a helium control gas line allowing not only the normal helium source gas to escape, but also the emergency jettison control gas supply as well (because of the failure of a check valve). Thus, the remaining residual propellants could not be jettisoned. The aircraft was an estimated 1500 pounds heavier than normal at landing but the landing was accomplished without incident.

The pilot performed a rudder pulse with the yaw damper off 71 seconds after engine shutdown and noted that the sideslip indicator did not oscillate as expected. Post-flight analysis of the maneuver revealed that the aircraft did in fact experience a reasonable yaw rate and lateral acceleration. The maneuver was performed at approximately the time of maximum temperature for the unprotected Ball Nose. It was concluded that the sphere of the Ball Nose experienced binding, possible due to differential expansion.

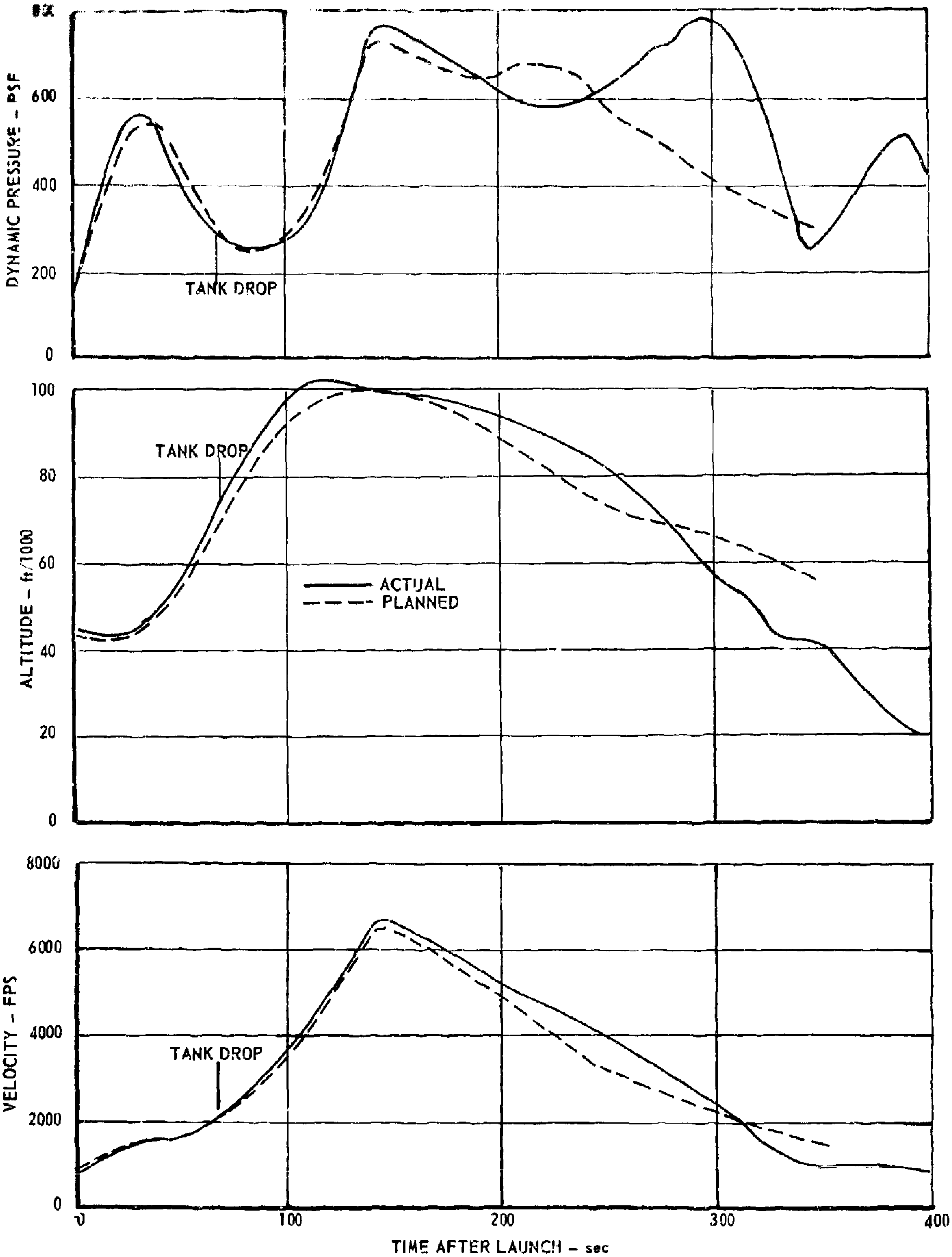


Figure 40 FLT. 2-53 PROFILE COMPARISON

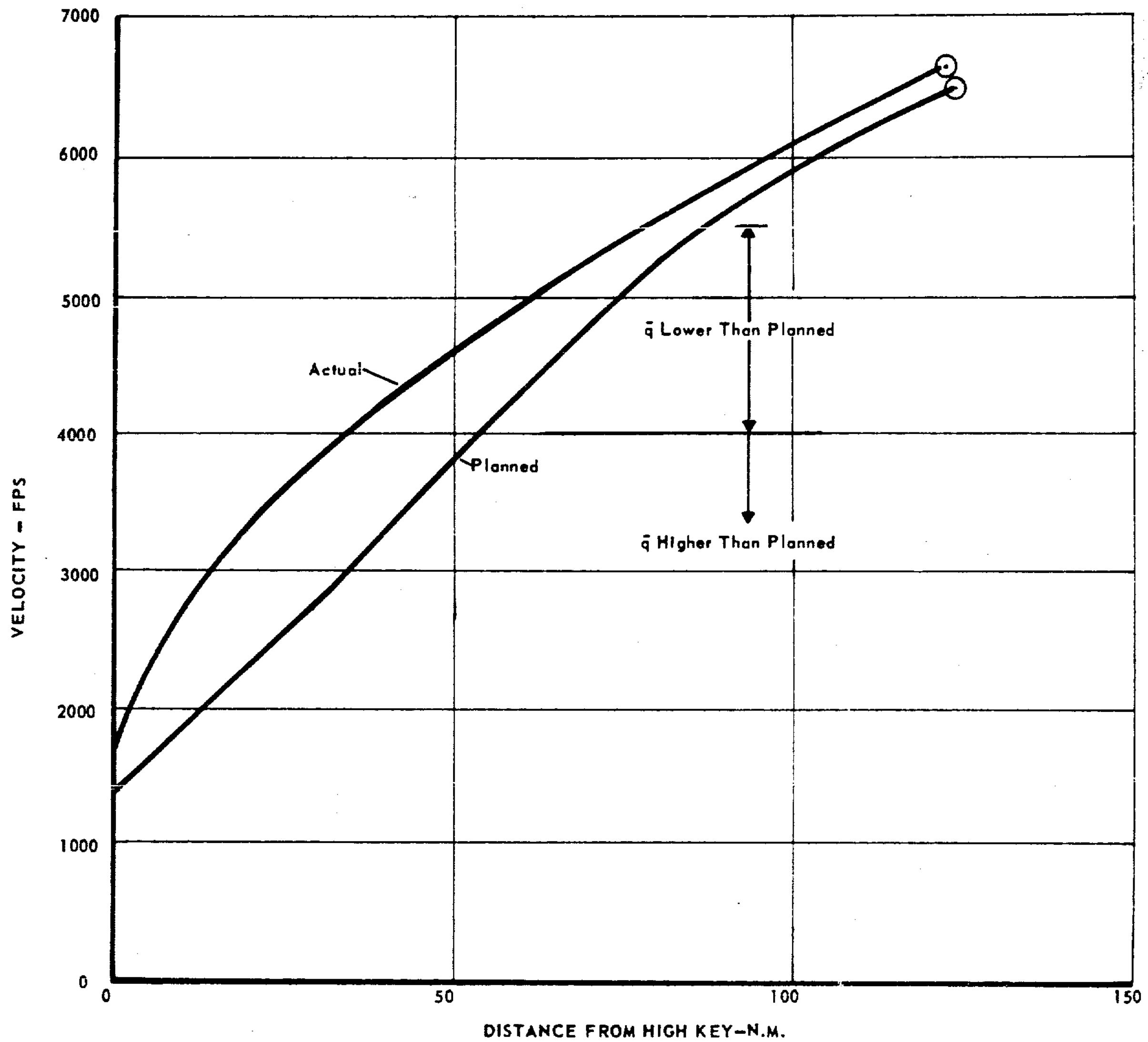


Figure 41 FLT. 2–53 TERMINAL RANGING

The heat in the ramjet pylon area became high enough to ignite 3 of the 4 explosive bolts retaining the ramjet to the pylon at some time during the flight. As the pilot was performing a turn to downwind in the landing pattern, the one remaining bolt failed structurally and the ramjet separated from the aircraft. The pilot did not feel the ramjet separate. Since the landing chase aircraft had not yet joined up, the pilot was not aware that the unit had separated. The conditions at the time the ramjet separated are shown in table VI.

The position of the aircraft at the time of separation was established by radar data and the most likely trajectory estimated. A ground search party discovered the ramjet impact point on the Edwards AFB bombing range. Although it had been damaged by impact, it was returned for study of the heat damage that had occurred.

TABLE VI

RAMJET SEPARATION CONDITIONS

FLIGHT NO. 2-53-97

Velocity	980 fps	Angle of attack	8^{o}
Altitude	35,5000 feet	Roll angle	57^{o} left
Mach Number	.98	Normal accel.	1.6 g
Dynamic Pressure	340 psf		

The unprotected right-hand windshield was, as anticipated, partially covered with ablation products. With the pilot's visibility being restricted (the left window was still covered by the eyelid) his guidance to the high key position was based on radar vectors from ground control. The eyelid was opened at approximately 1.6 Mach number as the aircraft was over Rogers Lake and the visibility out this window was good.

SUMMARY OF TEST RESULTS

External Tank Ejection and Recovery System Operation

A total of four in-flight ejections of external tanks were made from the aircraft during the envelope expansion program. Three of these were close to the planned ejection conditions and were within the original published boundary established for good separation characteristics (figure 42). The remaining ejection occurred during a preplanned alternate flight plan. The conditions at the time of the ejection were considered severe since the aircraft was outside the most optimistic limit of dynamic pressure/angle of attack and the tanks were partially full. The fact that the system survived the unexpected test without mishap provided additional confidence in the separation characteristics. A summary of the tank ejection conditions - both planned and actual - is presented in table VII.

All eight tanks had separated from the aircraft satisfactorily since no evidence was found that the tanks had ever recontacted the aircraft. Adequate instrumentation was never developed to determine the actual separation characteristics of the tanks. Ground based photographic coverage provided only qualitative information about the recovery system and was not adequate for detailed analysis of the tank motion with respect to the aircraft.

After flight 2-50-89, it was discovered that the nose-mounted separation rocket motor from the LOX tank had failed to fire. The igniter had fired giving the nozzle a used look; however, the solid propellant grain had failed to ignite. Further inspection of "spent" rocket cases uncovered

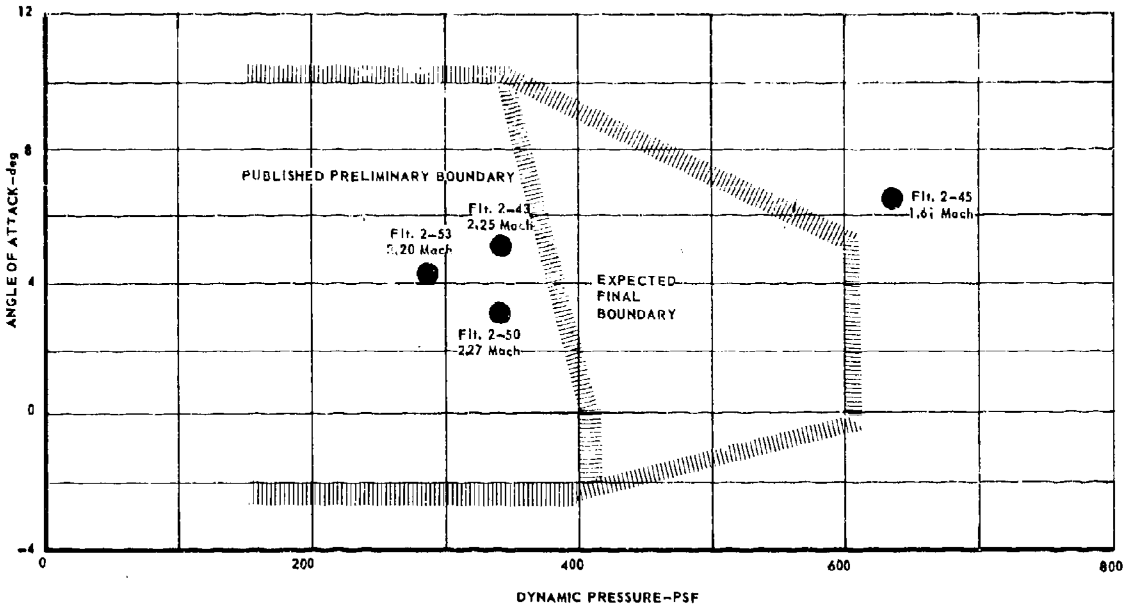


Figure 42 EXTERNAL TANK EJECTION CONDITIONS REFERENCED TO EJECTION LIMIT

Table VII

SUMMARY OF TANK EJECTION CONDITIONS							
Flight No.	Velocity (fps)	Altitude (ft)	Mach No.	Dynamic Pressure (psf)	α (deg)	Normal Acceleration (g)	Time From Engine Light (sec)
43 Plan 43 Act.	2,100 2,170	69,000 70,300	2.16 2.25	300 343	5 5.0	0.7 0.65	76 81.8
45 Plan 45 Act.	2,100 1,570	68,000 41,900	2.16 1.61	349 618	5 6.5	0.5 1.18	66 28.0
50 Plan 50 Act.	2,100 2,170	69,000 69,500	2.16 2.27	320 342	5 3.5	0.5 0.32	66 67.6
53 Plan 53 Act.	2,000 2,170	66,000 73,500	2.06 2.20	340 288	5 4.2	0.5 0.28	65 66.12

one additional rocket motor with the grain unburned. This second rocket was determined to be from the ammonia tank on flight 2-43-75. The effect of these failures on the separation characteristics of the tanks could not be determined because of lack of data, but it was assumed that the separation had been degraded without the nose down acceleration provided by the rocket. Although separation was apparently adequate on both ejections without the separation rocket, no serious consideration was given to eliminating it from the system since both of the ejections were at almost identical conditions and near the optimum. Modifications were made to the igniter of the rocket to eliminate other hangfires.

The tanks were unstable during free flight with the drogue chute. Although some changes were made to the system between flights, the tanks continued to exhibit an unstable motion. This tumbling action never pre-

sented a problem since the drag of the tumbling tank was large enough to decrease the velocity adequately before main chute deployment. The recovery system was adequate; all but one of the eight tanks were recovered in refurbishable condition. The LOX tank from flight 2-43-75 was destroyed on impact when an explosive bolt which secured the nose cone covering the main chute compartment failed.

Ablative Performance

The ability of the ablative coating to protect the aircraft structure from high temperature was better than expected; the maximum temperatures recorded on flight 2-53-97 were lower than post-flight calculations based on the actual flight conditions encountered (reference 6).

The condition of the ablative material after flight 2-53-97 was considered good and is shown in figures 43 through 49.

Figure 43 shows the charred leading edge surface of the wing and the condition of the underside of the fuselage. The effect of increased heating at the leading edge of the open external tank propellant access door is apparent. This door failed to close after ejection of the external tanks. Figure 44 is a close up view of the wear on the leading edge of the wing and shows increased char in certain locations which is assumed to be the result of high heating from the bow or tunnel side fairing shock wave.

The condition of the ablative coating of the empennage area is shown in figure 45. The large amount of charring of the speed brakes was due to the high heat which resulted from extending the speed brakes to 35 degrees shortly after maximum velocity was reached.

The effect of unusual local flow heating on the nose section is shown in figures 46 and 47. Two areas of uncharred ablator appear to radiate aft of the ball nose sideslip pressure ports. The "mud crack" appearance is characteristic of the material as a result of cooling after having been exposed to a particular temperature. It must therefore be concluded that the temperature of the smooth surface was lower than the adjacent material which cracked due to this local flow. The underside of the nose was subjected to higher temperatures and thus the entire surface was cracked. The ball nose assembly was replaced between ablative flights and it is apparent that the material applied to the unit had different characteristics than that of the original application due to the marked difference in appearance aft of the ball nose assembly. Flow discontinuity of the partially filled in BCS rocket cavity caused the increased heating indicated by the charring at this location on the lower side of the nose.

The condition of the canopy after the flight is shown in figures 48 and 49. The extent of the "fogging" on the unprotected left windshield due to redeposit of the ablation products may also be observed.

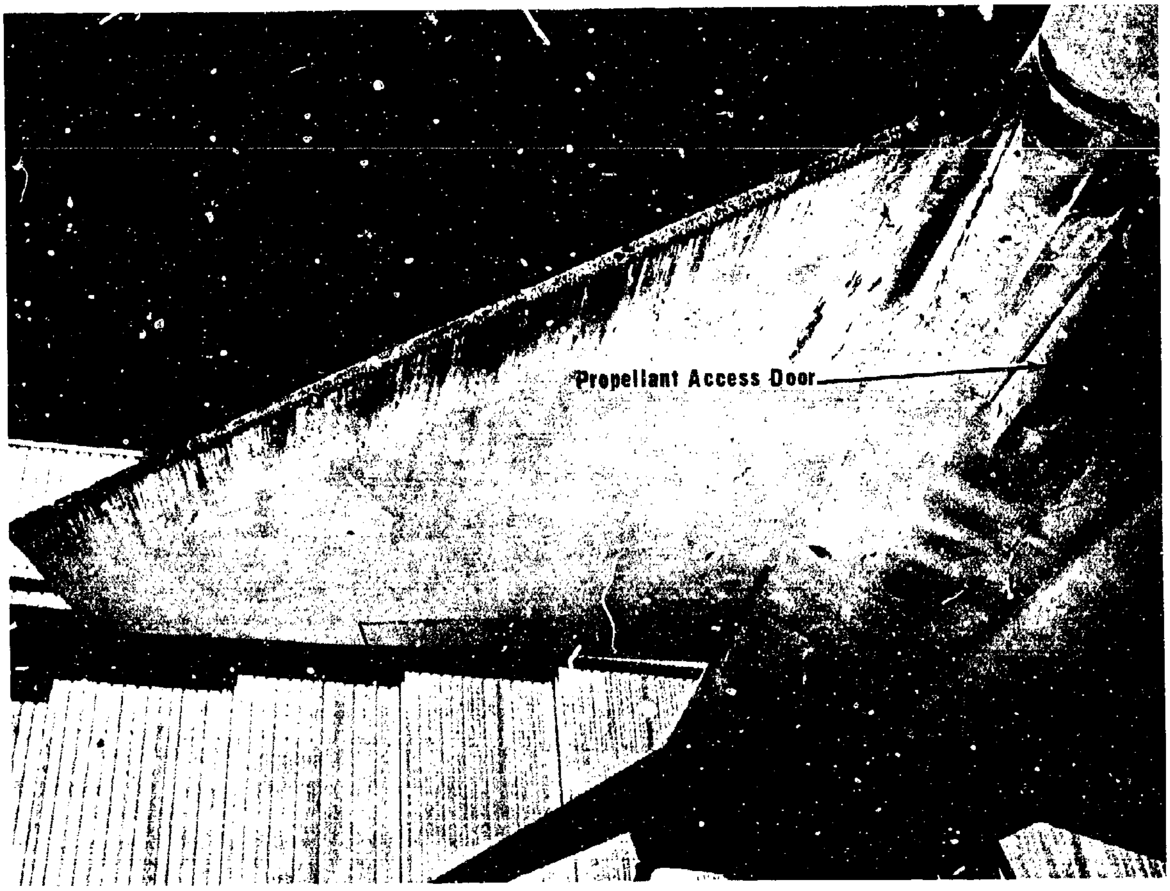


Figure 43. ABLATIVE WEAR LOWER FUSELAGE AND WING

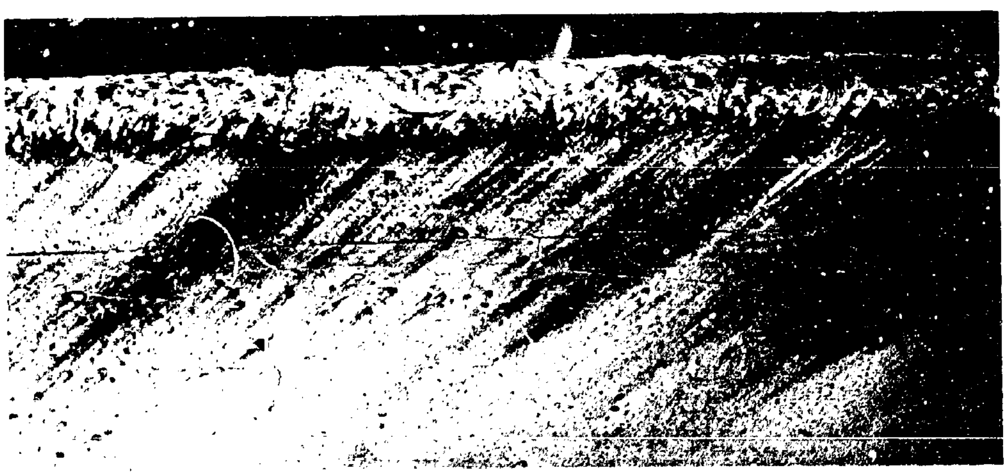


Figure No. 44 ABLATIVE WEAR WING LEADING EDGE

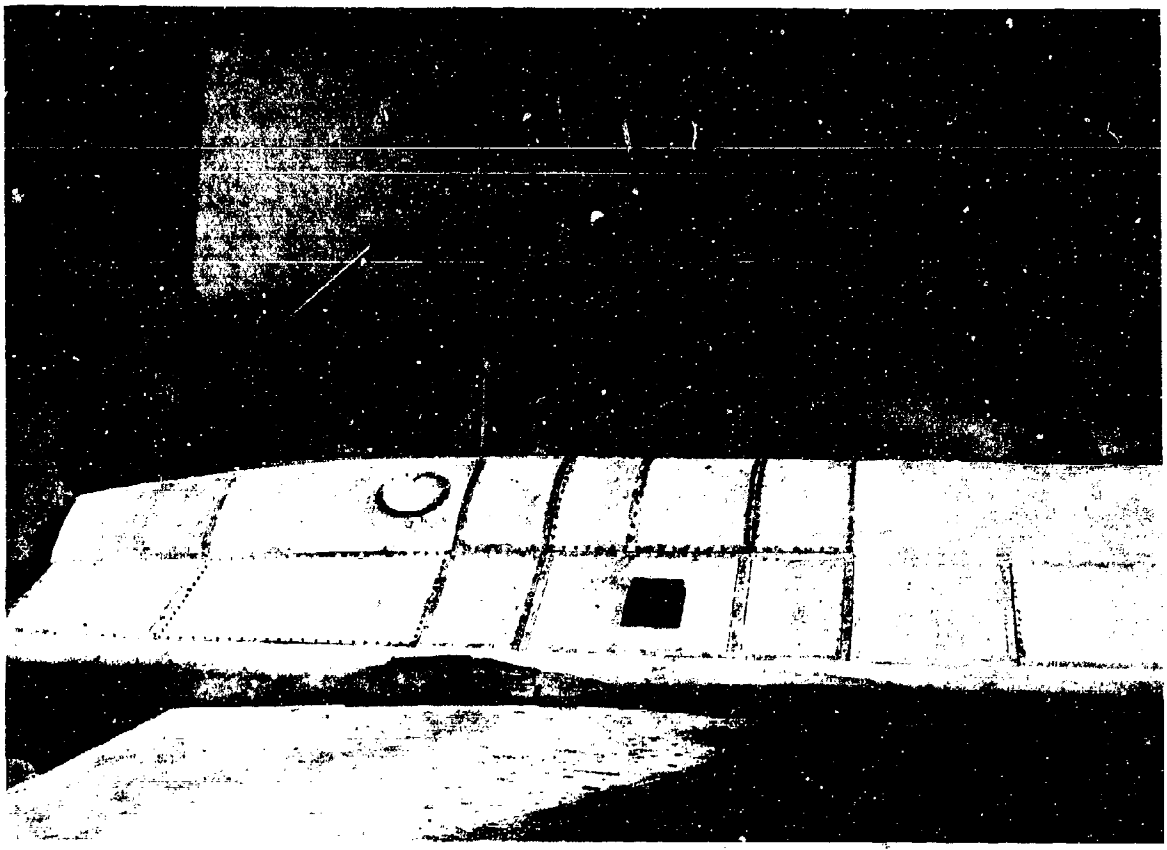


Figure 45 ABLATIVE WEAR ON EMPENNAGE

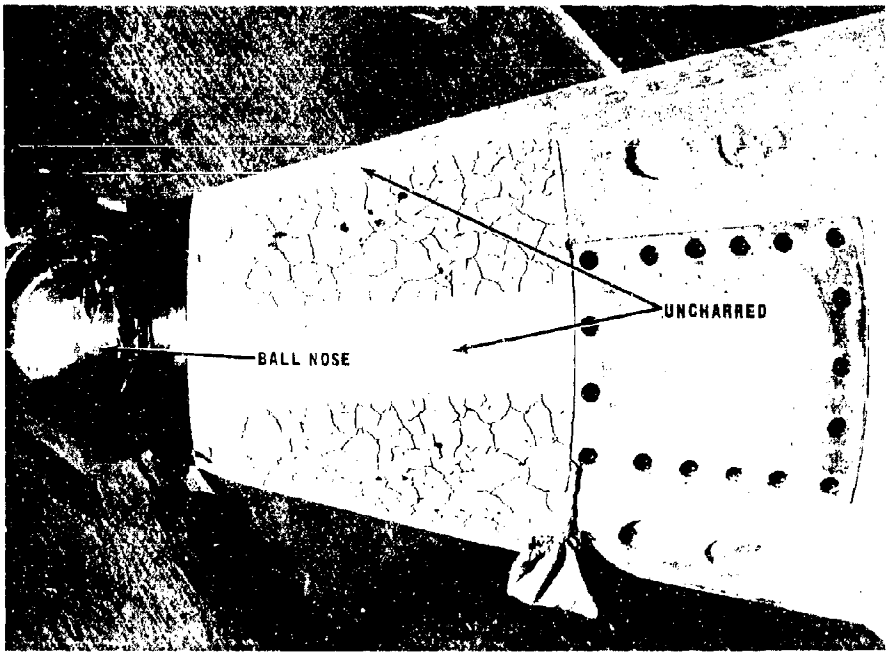


Figure 46 ABLATIVE WEAR UPPER NOSE

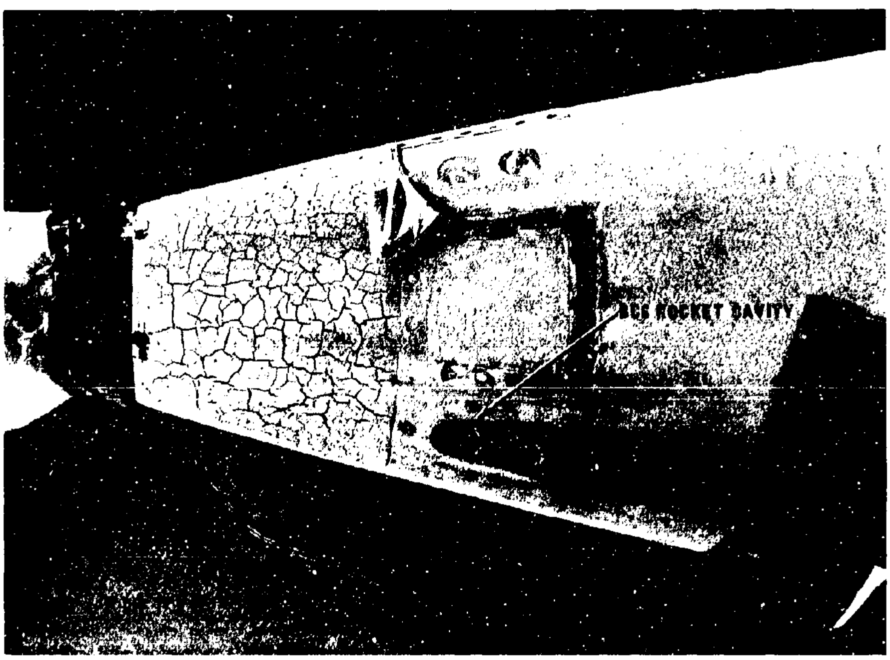


Figure 47 ABLATIVE WEAR LOWER NOSE

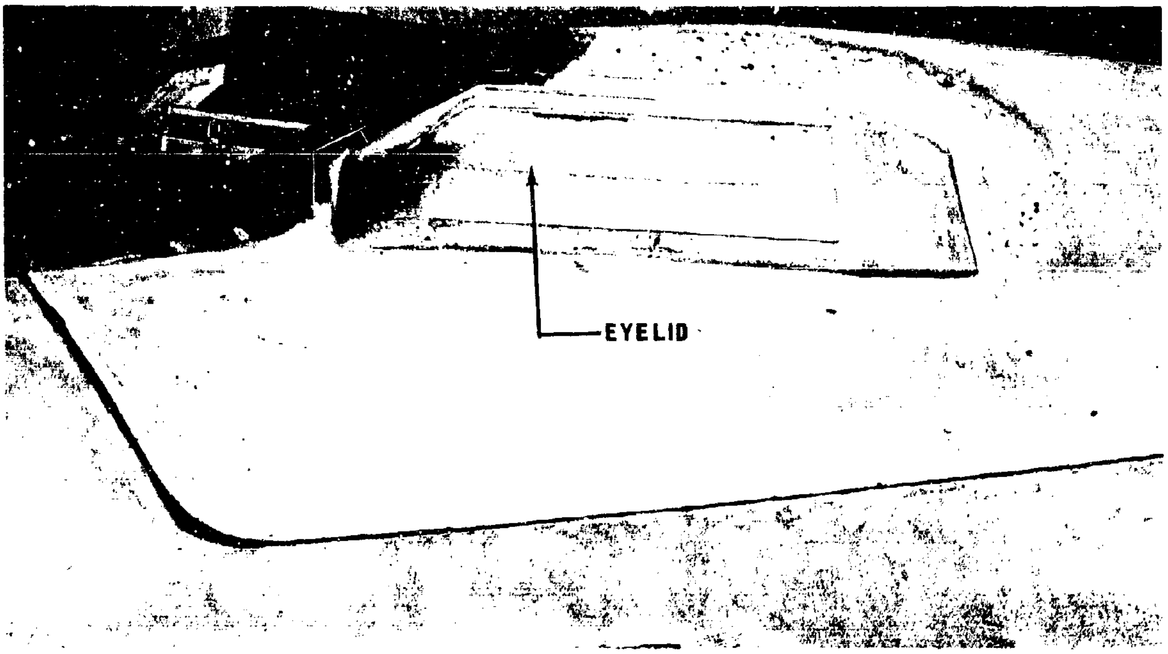


Figure 48 LEFT CANOPY

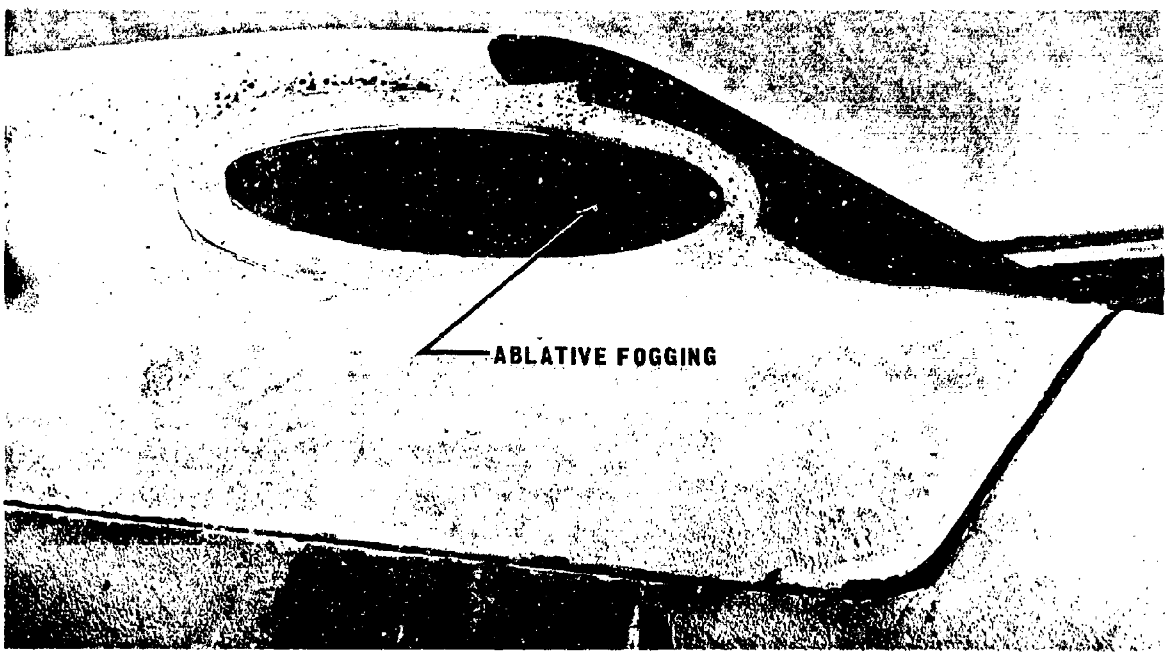


Figure 49 RIGHT CANOPY

Heating in the Ramjet Area

The severe structural damage to the dummy ramjet and pylon during the flight to Mach 6.7 was the result of local aerodynamic heating due to shock impingement and flow interference effects. Figure 50 shows the preflight condition of the ramjet, pylon, and lower fuselage.

The most severe melting damage occurred near the bottom of the ramjet pylon where shock waves generated by the ramjet spike tip, spike flare, cowl lip and bottom pressure probe were assumed to have intersected (figures 51, 52, and 53). A postflight thermal analysis of the heating in this area was made using the recorded temperature from the thermocouple located at the leading edge of the pylon and the observed heat damage as a guide. The measured temperature indicated a low value (less than 0°F) until approximately 145 seconds after launch when a rapid rise in temperature occurred, indicating that the ablative material had burned

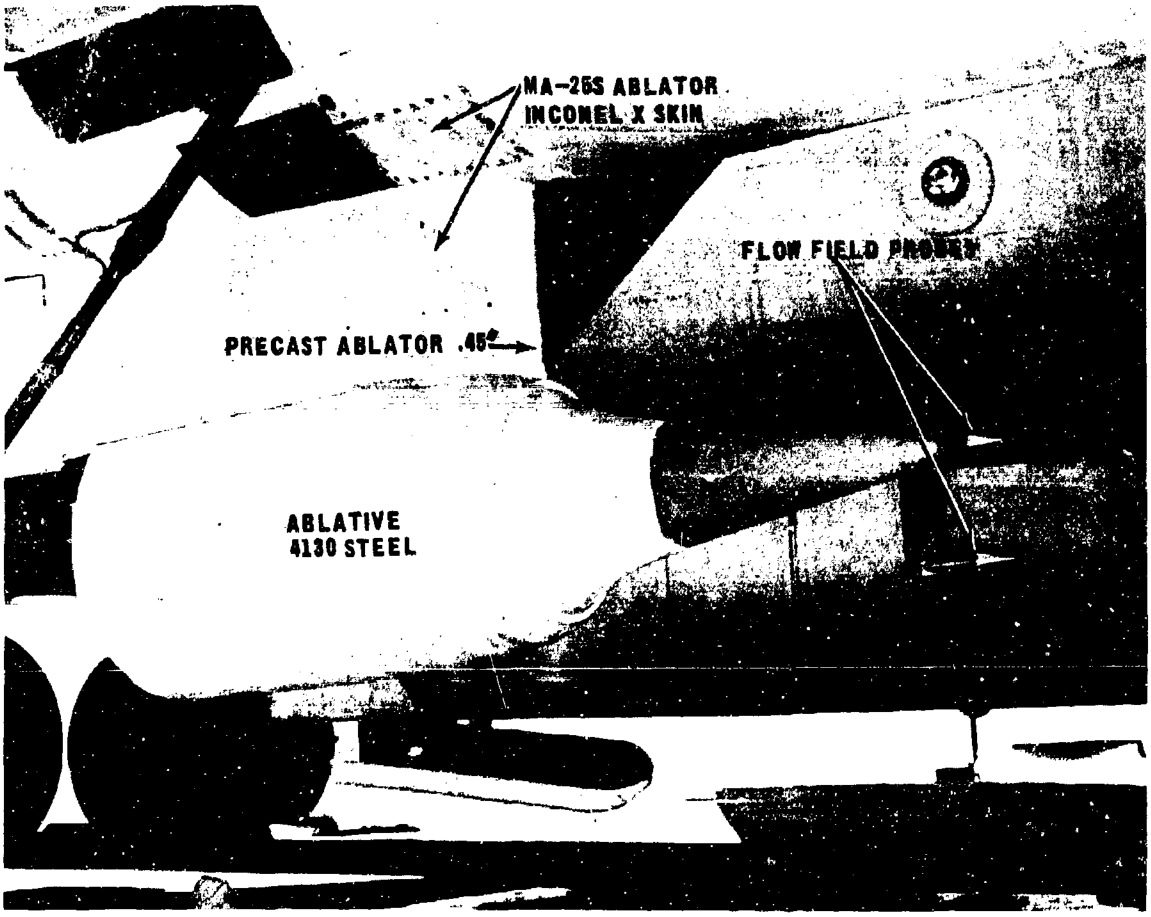


Figure 50 PRE-FLIGHT CONDITION OF RAMJET AND PYLON

through. The recorded temperature was increasing rapidly when the thermocouple wiring was severed by heat. A thermal analysis match of the ablator burn through time of 145 seconds was obtained when the undisturbed heat-transfer coefficient was increased by a significant factor (reference 7). This analysis showed that the temperature was sufficient to result in the melting damage of the Inconel X (melting temperature approximately 2600 degrees F) pylon structure.

Two areas of high heating due to flow interference were the pylon/fuselage junction and the ramjet cowl lip. A reasonable temperature profile for the indicated damage was also obtained by increasing the undisturbed heat-transfer coefficient by a significant factor. The calculated temperature due to interference heating of the ramjet cowl lip exceeded the melting point of the 4130 steel (2800 degrees F) for a short time causing the melting damage shown in figure 54. A complete analysis of this subject may be found in reference 7.

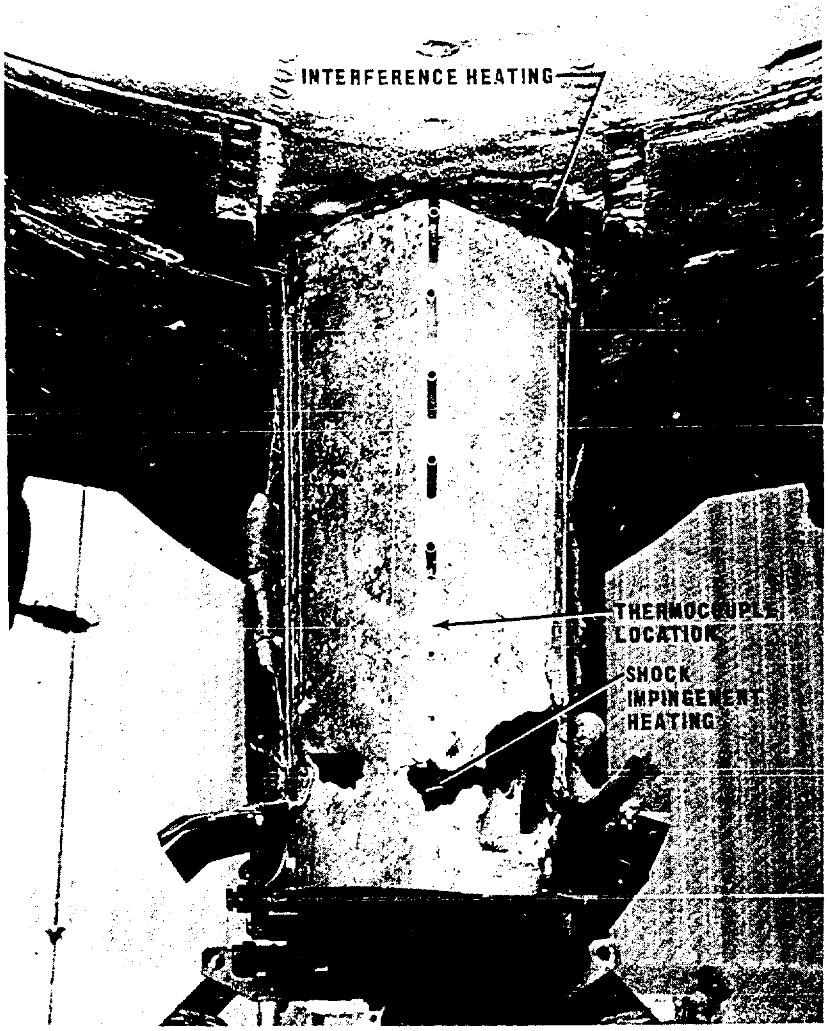


Figure 51 PYLON HEAT DAMAGE–FRONT

Figure 52 PYLON HEAT DAMAGE–LEFT SIDE

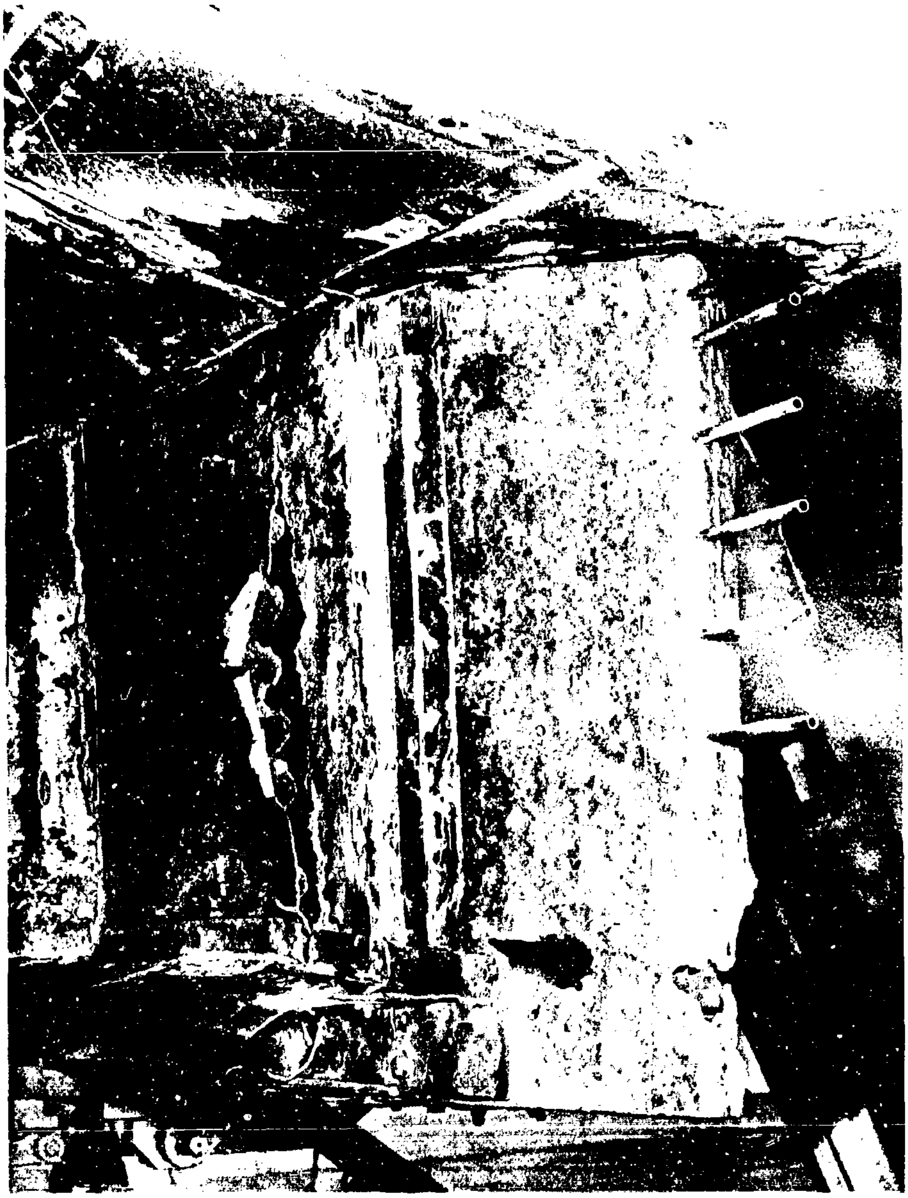


Figure 53 PYLON HEAT DAMAGE–RIGHT SIDE

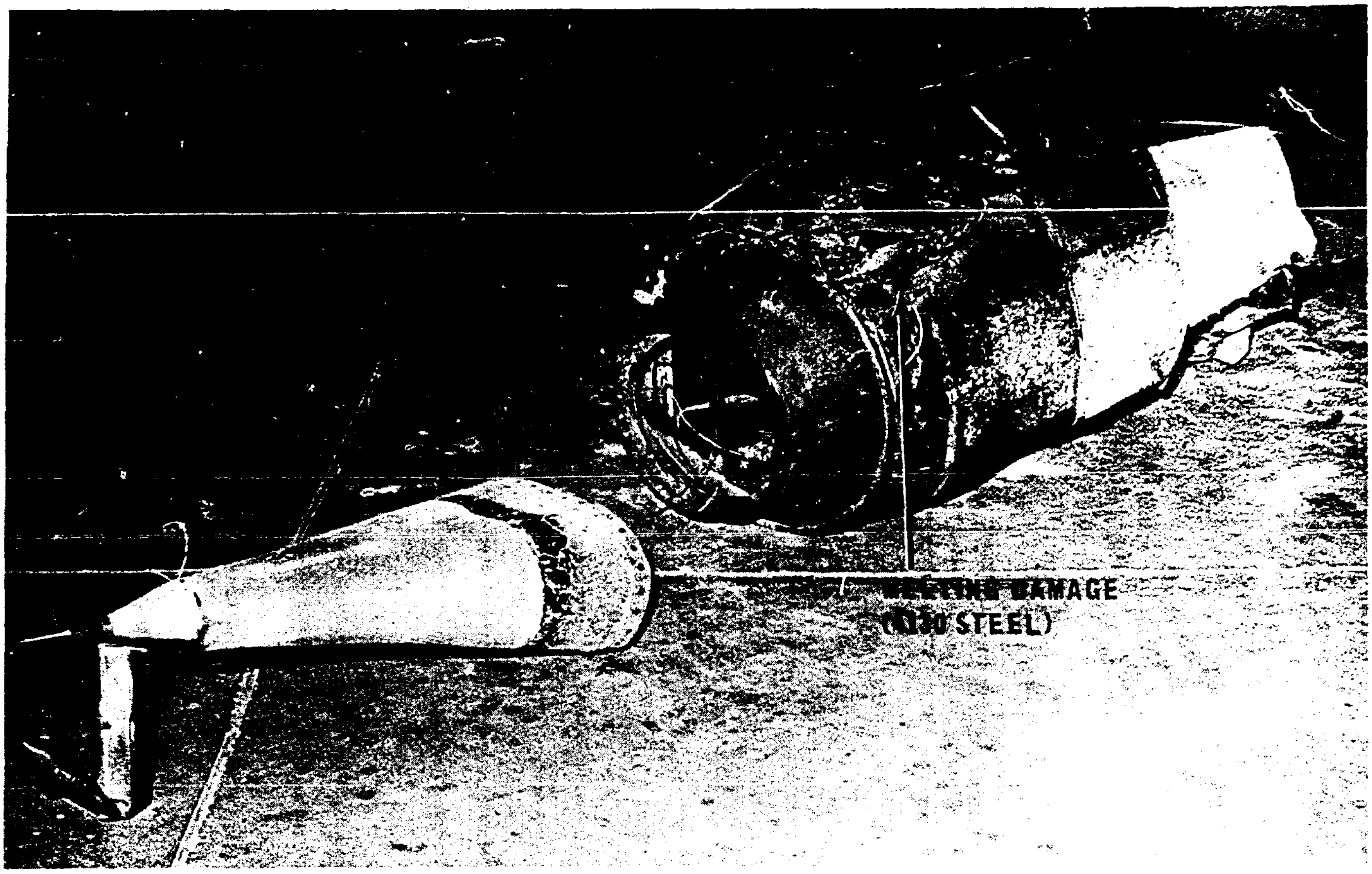


Figure 54 DUMMY RAMJET HEAT DAMAGE

Ablative Drag

An analysis was made of the X-15A-2 flight data to determine the effect of the ablative material on aircraft drag and L/D at subsonic speeds. The data analyzed were from flight 2-51-92 without ablative material and flight 2-52-96 with the full ablative coating. The dummy ramjet was installed on both flights and the only external configuration difference was the ablative material coating.

The lift and drag coefficients were determined by the accelerometer method. The measured angle of attack from the ball nose of the X-15 is subject to an error at subsonic speeds due to upwash effects. The measured angle of attack was corrected for this error by dividing the measured value by a factor 1.46. This factor was determined by comparing subsonic data from the ball nose with data obtained using standard nose boom vanes on early X-15 flights.

The resulting change in drag coefficient due to the ablative material is shown in figure 55. At a lift coefficient of 0.35 the increase in drag coefficient attributed to the ablative material is 15 percent (0.125 ΔC_D).

The change in L/D due to the ablative material, shown in figure 56, amounts to one-half an L/D decrease or 15-percent reduction at 0.35 C_L. It should be remembered that the ablative material on flight 2-52-96 was not significantly charred in comparison to that experienced on flight 2-53-97. A comparative analysis of L/D was not possible with flight 2-53-97 because of the premature separation of the ramjet and the external damage to the pylon. However, an additional degradation in L/D would be expected as a result of the additional roughness that occurred on flight 2-53-97 or for flight to even higher speeds with resulting increased charring.

Figure No. 55
X-15A-2 DRAG POLARS
WITH & WITHOUT ABLATIVE COATING
DUMMY RAMJET CONFIGURATION
DATA PTS. .48<MACH<.75

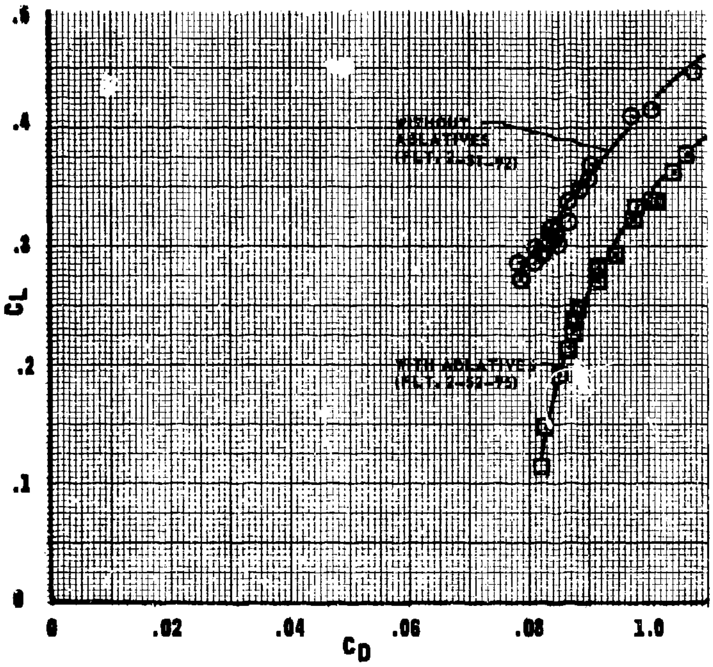
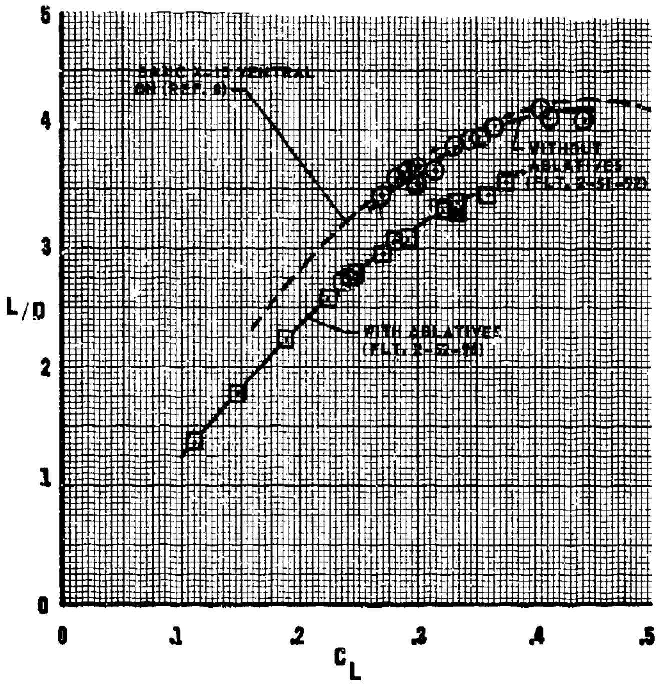


FIGURE No. 56

EFFECT OF ABLATIVES ON X-15A-2
L/D AT SUBSONIC SPEEDS
DUMMY RAMJET CONFIGURATION
DATA PTS. .48<MACH<.75



CONCLUSIONS

The redesigned propellant system, with external tanks to contain additional propellants for increased performance capability of the X-15A-2, was brought to maturity through ground test stand development and flight test. Adequate tank separation characteristics were demonstrated and the designed recoverable/refurbishable concept of the external tanks proven.

A satisfactory room-temperature cure ablative material was selected from several candidate materials through wind tunnel arc tests and through flight test in small quantities on the X-15 aircraft. With the limited test result from 2 flights with a full coating on the aircraft it appeared that a satisfactory ablative material had been developed to protect the aircraft structure from the high temperature associated with flight of the aircraft outside its original design envelope.

A real-time analog simulation of temperature resulting from aerodynamic heating was developed. This simulation was used in conjunction with the six degree of freedom simulation of the aircraft to plan flights in which temperature at particular locations were one of the constraints of the desired flight. This combined simulation was also utilized during pilot training for the flight to enable the pilot to become aware of the effect of off-design flight conditions on the resulting temperature.

During the course of the flight program of the modified X-15A-2 several in-flight failures occurred which dramatically demonstrated the effects of relatively minor changes in configuration of a vehicle operating in an environment where aerodynamic heating is significant. Thermal loads caused premature extension of one or more components of the modified landing gear system at high Mach number on three occasions.

Increased heating resulting from shock impingement and flow interference from the dummy ramjet installation caused severe structural damage to the ramjet and pylon.

The type of problems encountered during the course of the envelope expansion program may well be expected on other vehicles operating in the speed regime where aerodynamic heating will be an appreciable factor. In addition to the continued demonstration of piloted landing of an unpowered-low L/D vehicle, other techniques developed during the program are applicable to orbital lifting re-entry vehicles: application of and flight with an ablative coating, protection of a canopy window with a pilot-actuated covering, and development of an extendable pitot tube as an airspeed source for the terminal landing maneuver.

APPENDIX I

FLIGHT No. 2-53-97
FLIGHT PLAN

X-15 FLIGHT PLAN

Aug. 30, 1967.

Flight No.: 2-53-97 Scheduled Date: Sept. 19, 1967

Pilot: Major William J. Knight

Purpose:
1. Martin Ablative Test (Fullcoat) - Watts
2. Stability & Control with Dummy Ramjet - Robinson
3. Ramjet Local Flow Test - Nugent
4. External Tank Separation Characteristics - Bryant
5. Wing Tip Accelerometer - Kordes
6. Fluidic Temperature Probe - Webb

Launch: Mud Lake #1 on a magnetic heading of 180°. SAS Hi-Hi-Hi (8-6-8) YAR OFF, Pitch-Roll and Yaw ASAS Armed, Tank Eject "ARMED", Propellant Flow "EXTERNAL", Dummy Ramjet installed.

Launch Point Coordinates: 37° 58.5' N; 116° 50'W

Instrumentation Engineer: Paul Harney

Item	Time	Alt.	Vel.	α	$\bar{q}$	Event
1.	0	43.5	770	2	145	Launch, light engine at 100% T. Rotate at 12° α until θ = 35°. $\bar{q}$ max = 540 PSF. N_Z max = 1.8 g.
2.	41	48	1600	12	540	θ = 35°. Maintain θ = 35°.
3.	60	62	1900	6	360	Pushover to 5° α. Check external flow - switch to internal.
4.	65	66	2000	5	340	Eject external tanks. Maintain α = 2° after trim change.
5.	124	99	5300	2	460	Gradually increase $\alpha \approx 7°$ (N_Z = .7 g) to maintain H = 100,000 ft.
6.	141	100	6500	7	730	Shutdown. Pushover momentarily to 2° α, then pull up to 7° α(H ≈ -150 fps).

Flight No.: 2-53-97 Scheduled Date: Sept. 19, 1967

Item	Time	Alt.	Vel.	α	$\bar{q}$	Event
7.	152	99	6300	7	720	Dearm Yaw ASAS - Yaw Damper OFF - Perform δ_V pulse at 7° α - Yaw Damper Hi.
8.	158	98	6200	7	700	Extend speed brakes to 35°. Maintain $\alpha = 7°$.
9.	168	96	5900	7	680	Yaw Damper OFF - perform δ_V pulse at 7° α - Yaw Damper Hi.
10.	178	95	5500	7	660	Pitch damper Lo - Perform δ_H pulse - Pitch Damper Hi. Increase α to achieve $\dot{H} \approx -200$ fps.
11.	189	92	5200	8	640	Trim down to 2° α then back to 8° α ($\dot{H} \approx -300$ fps)
12.	208	86	4600	8	680	Yaw Damper OFF - Perform δ_V pulse at 8° α - Yaw Damper Hi. ($\dot{H} \approx -200$ fps).
13.	218	83	4200	8	680	Trim down to 2° α then back to 8° α ($\dot{H} \approx -300$ fps).
14.	236	77	3600	8	660	Yaw Damper OFF - Perform δ_V pulse at 8° α - Yaw Damper Hi. Maintain $\dot{H} = -200$ fps.
15.	245	74	3200	5	600	Retract speed brakes.
16.	275	69	2600	6	500	Yaw Damper OFF - Perform δ_V pulse - Yaw Damper Hi.
17.	283	68	2500	6	560	Open canopy eyelid. Yaw Damper OFF - Perform δ_V pulse - Yaw Damper Hi. Rearm Yaw ASAS. Vector to High Key.
18.	311	64	2000	5	380	Extend alternate pitot tube. Perform trim evaluation by maintaining $\alpha \approx 5°$ between Mach 2 and 1.5, if energy management permits.

Flight No.: 2-53-97 Scheduled Date: Sept. 19, 1967

Item	Time	Alt.	Vel.	α	$\bar{q}$	Event
19.	348	56	1500	5	300	High Key - check flap and squat circuit breakers in. Check ramjet Armed. Engine Master OFF.
20.						Final approach - jettison ramjet.

NOTES:

1. θ vernier will be set at 35°, α crosspointer will null at 5°. Precision heading will be set at 180°.

2. Emergency Lakes: Grapevine, Ballarat, Cuddeback.

3. Flight Duration: Approximately 9 minutes.

4. Flight plan based on 60,000# thrust at 100%. Engine 110.
   Minimum thrust = 35,000#.
   Total burn time at 100% T = 144 secs.
   External = 57 sec. Internal = 87 sec.

Configuration	Weight-lbs	C.G. = %
Launch	52,117	
External Tank Depletion	39,205	
Ejected Tanks	2,224	
After Tank Ejection	36,982	
Shutdown	17,967	
Burnout	17,288	
Inflight Jettison Check		

5. ΔC_D = .015 below 5,500 fps. (Allowance for ablative drag.)
   ΔC_D = .02 above 5,500 fps.

GROUND RULES FOR NO LAUNCH:

1. Radio, Radar, or TM malfunction.

2. Malfunction of any SAS or ASAS channel.

3. Malfunction of Inertial Platform.

4. Malfunction of Ball Nose.

5. No external propellant flow.

Flight No.: 2-53-97 Scheduled Date: Sept. 19, 1967

GROUND RULES FOR NO LAUNCH, cont'd:

6. Loss of any ablative coating.

7. Malfunction of alternate attitude indicator.

ALTERNATE SITUATIONS AFTER LAUNCH:

Alternate profile (Tanks off):
Rotate at 2.4 g until $\theta = 35^\circ$. Maintain $\theta = 35^\circ$ to 54,000 feet, then pushover to zero g, shutdown at 5400 fps. (Peak altitude = 100-110 K).

Failure	Action
1. Radio or Radar	Proceed as planned.
2. Total Pitch Damper	Proceed as planned, shutdown at 5400 fps (127 sec.). Do not extend speed brakes until required for terminal energy management.
3. Attitude Failure	Use 7° α instead of 35° θ at 41 sec. Shutdown at 5400 fps.
4. Ball Nose Failure	Perform rotation by trimming stabilizer to -3° until $N_Z = 1.8$ g then maintain 1.8 g until $\theta = 35^\circ$. Use 65 sec. for tank ejection with cross checks on IAS (382 kts). Use 0.5 g for tank ejection. Shutdown at 5400 fps.
5. Delayed Engine Light	After first light attempt, eject external tanks "FULL" and proceed with alternate profile, pushover to 54,000 ft. (Burnout at ≈ 4900 fps.)
6. External Flow Failure	0 - 10 sec. Status Check 10 - 52 sec. Shutdown, eject tanks "PARTIAL", relight engine and proceed with alternate profile. 52 - UP sec. Eject tanks "EMPTY" and proceed with normal profile. (Use "FULL" button if flow fails immediately after launch.)

Flight No.: 2-53-97 Scheduled Date: Sept. 19, 1967

ALTERNATE SITUATIONS AFTER LAUNCH, cont'd:

Failure	Action
7. Premature engine shutdown.	After shutdown with tanks on - eject tanks as per item 6. Attempt relight, if successful proceed with alternate profile. No engine light - proceed to emergency lake.

Planned Profile

0 - 44 sec	Mud Lake	
		(1550 fps)
44 - 94 sec	Grapevine	
		(3200 fps)
94 - 107 "	Ballarat	
		(4000 fps)
107 - 115 "	Cuddeback	
		(4500 fps)
115 - UP	Edwards	

Alternate Profile (Delayed engine light - Tanks off)

0 - 43 sec	Mud Lake	
		(2100 fps)
43 - 68 "	Grapevine	
		(3400 fps)
68 - 80 "	Ballarat	
		(4300 fps)
80 - 83 "	Cuddeback	
		(4500 fps)
83 - UP	Edwards	

8. Failure of empty tanks to separate at planned tank drop point (65 sec): Start turn to Mud Lake, shutdown at Vel. < 2500 fps. Relight engine when required near completion of turn and proceed to Mud Lake at minimum thrust. Shutdown at approximately 2000 fps at 50,000 feet and vector to Mud Lake while jettisoning internal propellants. If aircraft control appears satisfactory, a landing may be considered at pilot's discretion.

9. Failure of tanks to eject with external propellants remaining

No landing.

Approved by: J Adkins

E. J. Adkins, Chief
X-15 Research Project Office

X-15 FLIGHT PLAN REVISION

Sept. 25, 1967

Flight No.: 2-53-97

Make the following pen changes to flight plan dated Aug. 30, 1967.

ALTERNATE SITUATIONS AFTER LAUNCH:

Page 4:

Change Item 2 from "Total Pitch Damper" to "Any Damper Failure."

Add Item 3a: Failure of inertials - proceed as planned: for velocity failure shutdown at 139 seconds.

Change Item 5: Delayed Engine Light -

If first engine light attempt is unsuccessful, one relight attempt may be tried. If successful proceed as planned, do not exceed 1000 PSF $\bar{q}$ or 2.4 g during rotation. Delay tank ejection until $\bar{q} < 400$ PSF.

No engine relight; eject external tank "FULL" at $\alpha < 10°$ Jettison propellants and recycle engine with throttle and ignitor OFF to establish pump idle flow. Proceed to Mud Lake.

Approved by [signature]
E. J. Adkins, Chief,
X-15 Research Projects Office

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13. ABSTRACT

After having been extensively damaged during an emergency landing on its thirty-first flight (November 9, 1962), the X-15-2 aircraft was rebuilt and modifications were incorporated to increase the vehicle performance capability to allow flight testing of a hypersonic ramjet engine. The increased performance was derived from additional propellants contained in two external drop tanks. The modified propellant system with the external tanks was satisfactorily developed on a ground test stand and performed adequately during flight. The ablative material developed to protect the aircraft against temperatures exceeding the original aircraft design appeared to perform satisfactorily on the two fully coated flights flown. On the last flight of this aircraft, the vehicle achieved a maximum Mach number of 6.7. Extensive heat damage was encountered on the dummy ramjet and lower ventral fin as a result of unexpected increased heating rates due to shock impingement and flow interference effects. While the aircraft was being reparied, the X-15A-2 program was terminated and the maximum speed capability of the aircraft was never achieved. The type of problems encountered during the course of the envelope expansion program may well be expected on other vehicles operating in the speed regime where aerodynamic heating will be an appreciable factor. In addition to the continued demonstration of piloted landing of an unpowered low L/D vehicle, other techniques developed during the program are applicable to orbital lifting re-entry vehicles: application of and flight with an ablative coating, protection of a canopy window with a pilot-actuated covering, and development of an extendable pitot tube as an airspeed source for the terminal landing maneuver.

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envelope expansion program						
external drop tanks						
ablative material						
dummy ramjet						
aerodynamic heating						
flow interference effect						
canopy window eyelid						
extendable pitot tube						

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